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SPACE SHUTTLE
ORBIT MANEUVERING ENGINE
REUSABLE THRUST CHAMBER

NAS9-12802)

FINAL DATA DUMP

CASE FILE
COPY

Rocketdyne
North American Rockwell

SPACE SHUTTLE

ORBIT MANEUVERING ENGINE

REUSABLE THRUST CHAMBER

NAS9-12802)

FINAL DATA DUMP

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(NAS9-12802)

FINAL DATA DUMP

Prepared For
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Houston, Texas

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FOREWORD

This data dump, prepared in accordance with G. O. 07634, is submitted in compliance with Contract NAS9-12802, Exhibit A (Statement of Work), Task I (Reusable OME Thrust Chamber Evaluation and Task II (Alternate OME Propellant Combinations).

ABSTRACT

Presented is a data dump containing Space Shuttle Orbiter Maneuvering Engine performance, weight, envelope, and interface pressure requirements for candidate propellant combinations (NTO/MMH, NTO50-50, LOX/MMH, LOX/50-50, LOX/N_2H_4, LOX/C_3H_8, and LOX/RP-1) and cooling concepts (regenerative and dump/film). These data are presented parametrically for the thrust, chamber pressure, nozzle expansion ratio, and engine mixture ratio ranges of interest. Also included is information describing sensitivity to system changes; reliability, maintainability and safety; development programs and associated critical technology areas; engine cost comparisons during development and operation; and ecological effects.

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INTRODUCTION AND SUMMARY

The requirements for the space shuttle vehicle dictate an orbital maneuvering propulsion system which is lightweight, exhibits high performance, is insensitive to mission duty cycle, has long life, and is reusable, inspectable, and maintainable. Selection of the optimum propellant combination and engine cooling mode for this system requires the proper compromise between cost, performance, and degree of system sophistication.

Preliminary NASA, vehicle contractor, and propulsion contract studies have identified several propellant combinations and engine cooling modes for consideration in the Orbital Maneuvering System (OMS). Parametric data have been generated for the propellant combinations:

NTO/MMH
NTO/50-50
LOX/MMH
LOX/50-50*
LOX/N_2H_4**
LOX/RP-1
LOX/C_3H_8 (liquid)**

and engine cooling modes

Regenerative
Film

over a range of thrusts, chamber pressures, expansion ratios, and engine mixture ratios applicable to an Orbital Maneuvering Engine (OME). Table 1 summarizes the parametric design range, propellant combinations, and cooling modes encompassed by this data dump. Engine performance, weight, envelope, and interface pressure data were generated for each of the design points.

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* abbreviated parametric range

** data carry-over from Preliminary Data Dump (Ref. 1)

In addition to the parametric data the cooling modes and propellant combinations were evaluated relative to system sensitivity; reliability, maintainability, and safety; required development programs and the associated critical technology areas; cost both during development and engine operation; and the effect of engine exhaust gas in the environment.

POINT DESIGNS

Regenerative and film cooled engines were designed for pressure fed orbital maneuvering engines for the Space Shuttle Orbiter Vehicle. Layout drawings of these engine assemblies are shown in Figs. 1 and 2 . The engine components include propellant ducting with flex joints for gimbaling, a quad-redundant bipropellant valve, injector, thrust chamber, nozzle extension, and throat gimbal ring.

Propellant ducting carries the propellants from the vehicle interface on the gimbal ring to the valves and then to the injector and chamber. The valve mounting orientation which minimized duct weight was selected for each chamber cooling method: perpendicular to the chamber axis for the regeneratively cooled chamber; parallel to the axis for the film cooled chamber.

The method of attaching components is based on tradeoffs between reliability, maintainability, and weight. Components of lower reliability, such as the valve, are bolted onto the assembly to provide easy access for maintenance and repair. The injector and chamber are anticipated to have higher reliabilities and are, therefore, welded together to reduce engine weight. This configuration is also more reliable since it eliminates two injector-to-chamber seals (the fuel-to-hot gas seal and the fuel-to-ambient seal). If either the injector or chamber is damaged, they can be removed by machining to save the undamaged part.

All propellant ducts, the nozzle skirt, the propellant valves, valve actuation system, and the gimbal ring are bolted assemblies in the regenerative cooled engine. Additional weight savings could be effected by welding these components, but the gains do not appear sufficiently attractive at this time. The film cooled engine nozzle is welded to an integral part of the combustor (both columbium).

The OME flightweight valve is a quad-redundant valve (four valves in a series-parallel arrangement) to provide redundancy with either a failed-open or failed-closed valve. Individual oxidizer and fuel valves are mechanically linked to ensure mixture ratio control should a single valve malfunction. Each oxidizer-fuel valve combination is operated pneumatically by its own actuator and pilot valve.

A ball-type closure element provides low actuation force requirement and, therefore, low valve assembly weight. A low pressure drop in the valve ensures that the engine will operate within specified limits if a valve malfunction occurs. Failure of one fuel valve (an extremely remote possibility because it implies failure of a shaft or spline connection), would result in a 1 psi reduction in chamber pressure (2-percent thrust reduction) and a 4-percent increase in propellant mixture ratio if a bipropellant path fails to open, the mixture ratio would not change appreciably, although the thrust would decrease by approximately 2.5 percent. Loss of electrical signal to any valve will result in closing of the valve by pneumatic and spring forces. The valve will, therefore, continue to operate with an increase in ΔP of approximately 5 psi if electrical or pneumatic control of one side is lost.

Injectors with the appropriate type elements are used for each propellant combination. Unlike doublet elements are used for the NTO/amine propellant combinations. Like doublet elements are used with LOX/amines and LOX/RP-1. Coaxial element orifices are used with the LOX/C_3H_8 propellant combination. Oxidizer enters the center of the injector where, in the cases where LOX is the oxidizer, provision is made for an electrical spark ignition system. The injectors have provisions for acoustic cavities to suppress combustion instabilities.

Thrust is transmitted through the gimbal ring to the thrust mount pickup plates. These plates also serve as locators for the inlet ducts, electrical connectors, and for instrumentation connections. The gimbal assembly

is located at the thrust chamber throat plane and provides the capability of gimbaling the engine through angles of $\pm$ 7 degrees minimum in both the pitch and yaw axes. The assembly consists of a titanium circular ring into which self-aligning spherical bearings are installed at each of the four pivot points. The gimbal ring is a hollow rectangle in cross section for high bending and torsional resistance.

The OME thruster consists of a regenerative or film cooled combustor to which is attached a radiation cooled nozzle to reduce weight and coolant requirements. The relatively cooled regenerative chamber facilitates bolting the radiation cooled nozzle to the combustor. The nozzle is welded to, or on an integral part of, the film cooled chamber. A contraction area ratio of 2 was selected for both combustors based on considerations of cooling requirements, weight, performance, and combustion stability.

Uppass cooling was selected for the regeneratively cooled chamber to simplify the ducting and to minimize volume. The regeneratively cooled portion of the thrust chambers utilize channel wall construction with an electroform nickel close-out on the outer wall. A constant 0.030 inch hot wall thickness was selected as a near-minimum fabrication wall thickness although small benefits can be obtained by thickening the wall in some regions of the chambers to reduce the heat flux. Combustion chamber lengths were selected as approximately optimum for each of the conditions analyzed.

Small amounts of film cooling were used to supplement the regenerative cooling for MMH and 50-50 fuels. This supplemental cooling allowed lengthening of the combustor to improve combustion efficiency and reduced the regenerative coolant jacket pressure drop. A typical hot wall temperature profile is shown in Fig. 3 for an NTO/MMH regenerative cooled OME.

Supplemental film coolant was not used with LOX/C_3H_8, LOX/RP-1 and LOX/N_2H_4 because of the high decomposition temperatures for propane and RP-1 and because of the low mixture ratio for LOX/N_2H_4.

The dump/film cooled engine is shown in Fig. 2 . The combustion chamber, and nozzle, to an area ratio of 3, is insulated to maintain an outer wall temperature of 600F. Additional insulation to shield the gimbal bearing and propellant ducts from the radiation cooled nozzle is not shown in the drawing and has been estimated to weigh less than 1 lb. A detail of the injector end of the dump/film cooled chamber is shown in Fig. 4 . Some of the fuel entering the injector manifold is diverted to the chamber through the sleeve and dumped into the combustion chamber. The short sleeve is used to distribute and direct the flow of the coolant along the wall to provide essentially 100 percent cooling efficiency. The liner for the dump/film cooled combustion chamber is press-fitted into the shell and utilizes channels which are constant width and constant depth. The channels can be spiralled to some extent to improve distribution. A typical hot wall temperature profile is shown in Fig.5.

Corrosion resistant steel was used for the valves, injectors, and regeneratively cooled chamber liner (except the O_2/C_3H_8 chambers which used a copper liner because of its good propellant compatibility). The higher strength of INCO 625 was used to prevent buckling of the liners in the film cooled engines. Electroformed nickel was the closeout material on regeneratively cooled chambers. Silicide coated Columbium was used for the film cooled chamber and for all nozzles in the region of high heat flux. At lower heat fluxes (higher area ratios) titanium was used to reduce weight.

Life capabilities for the regeneratively cooled and dump/film cooled thrust chambers were determined employing the life analysis approach described in the Appendix. The difference in basic operation between these two cooling modes results in each concept being limited by a different damage mechanism.

Predicted creep damage fraction values for the regeneratively cooled chamber are shown in Fig. 6 . The small values of the creep damage fraction even with the safety factor of 4 indicate that creep is not a significant failure mode on the regeneratively cooled chamber. Predicted fatigue life for a typical channel wall chamber is shown in Fig. 7 . Under most severe operating conditions the CRES liners of the regeneratively cooled chambers are predicted to have a fatigue life of approximately 7000 cycles which, with a safety factor of 4, still exceeds the 1000 cycle requirement. Fatigue life for the copper channel wall chamber (O_2/C_3H_8) is much higher than for CRES.

Fatigue damage analysis for the dump film cooled chamber indicates that the fatigue damage fraction is very small, on the order of 5 percent. The more significant life limiting factor is creep and results because of the high thermal axial gradient along the shell in the region where the film coolant decomposes as shown in Fig. 8 . This condition results in relatively high stresses in the high temperature region of the chamber. The creep analysis for this set of conditions indicated a 75 percent damage fraction in the 15 hour required operating time including a safety factor of 4. This analysis was conservative in that: (1) no stress relaxation was allowed; and (2) the location of the high-temperature transient region will vary with time during the early portions of each firing so that creep damage will not occur all at one point as assumed in this analysis.

OME point design characteristics for the various propellants and cooling methods are listed in Table 2 for engines with a 50 inch static external exit diameter. Mixture ratios for engines using NTO/MMH and NTO/50-50 are based on equal propellant tank volumes. For the other propellants the mixture ratio is that which yields maximum delivered specific impulse. Engine lengths are measured from the top of the oxidizer duct to the end of the radiation cooled nozzle; combustor lengths from the injector

face to the throat plane. Engine inlet pressures are the pressures at the vehicle connect point and include the pressure drops through all engine components. Pressure drops for the cooling jacket and the injector are also shown in Table 2.

The regeneratively cooled engines deliver higher specific impulse performance than the film cooled engines and have higher engine weights and fuel inlet pressures. The highest performance is delivered by the O_2/C_3H_8 engine followed by O_2/MMH, O_2/50-50, and O_2/N_2H_4 which are approximately equal. The performance of O_2/RP-1 is about midway between the previously mentioned group and that of the NTO/amine group.

The effects of specific impulse, engine weights, and inlet pressure on loaded weight were determined using the OMS weights given in Ref. 2 and the trade fac tors given in Ref. 3. The results of the cooling method comparison shown in Table 3 indicate that the regeneratively cooled engine results in a 540 pound lower weight for the NTO/MMH OMS, a 1136 pound lower weight for the NTO/50-50 OMS, and a 712 pound lower weight for the LOX/MMH OMS than the film cooled engine. The results of the propellant comparison for regeneratively cooled OME systems is also shown in Table 3. NTO/50-50 yields a slightly lower system weight than NTO/MMH. The lowest system weight is achieved with LOX/MMH.

PARAMETRIC DATA

GENERAL GROUND RULES AND ASSUMPTIONS

The ground rules and assumptions employed in generating parametric data for the candidate OME engine configurations are summarized in Table 4. The thrust chamber consists of the combustion chamber and nozzle. The combustion chamber and lower area ratios of the nozzle are cooled regeneratively or with dump/film cooling. In some cases supplementary film cooling is utilized to reduce the bulk temperature rise of the regenerative coolant. The higher area ratio portions of the nozzle are radiation cooled. The nominal nozzle length is 70 percent of the length of the equivalent 15 degree half angle cone nozzle.

Generally, all engine configurations include a throat gimbal ring capable of 7 degree gimbaling, although the effects of no gimbaling and head-end gimbaling on weight and envelope were evaluated. Series/parallel propellant valves are used to provide fail operational/fail safe redundancy. Propellant ducting from the vehicle connect point to the valves and from the valves to the thrust chamber or injector are included. The engines are designed to operate for a minimum of 15 hours and 1000 cycles with safety factors of 4 on each of these design parameters.

Regenerative Cooled Chambers

The ground rules applied to the regeneratively cooled chamber designs are shown in Table 5 . The regenerative coolant inlet area ratio was based on radiation cooling requirements shown in Fig. 9 . For most propellant combinations a coated columbium nozzle operating at a maximum temperature of 2400F was employed. At the point at which the radiation temperature dropped below 1600F the nozzle material was changed to titanium to save weight. For the propane cooled nozzle, the additional

requirement that all of the propane be vaporized in the low heat flux (< 1 Btu/in^2-sec) region of the regenerative nozzle resulted in a relatively long regenerative nozzle and short columbium nozzle. Therefore, the regeneratively cooled nozzle was extended slightly so that a titanium nozzle could be attached directly.

Maximum coolant bulk temperatures for 50-50 and MMH were based on maintaining a 50F subcooling, referred to chamber pressure. For N_2H_4 the maximum bulk temperature allowed was 280F. The film coolant flows were based on providing an engine design which could operate with the specified bulk temperature limits at $\pm$ 10 percent chamber pressure and $\pm$ 12 percent mixture ratio. Combuster length and auxiliary film coolant flow (where required) were optimized to provide maximum performance consistant with temperature and life criteria.

Coolant bypass circuits were used for both the propane and RP-1 propellant combinations. Bypass is required for the propane in order to keep the inlet area ratio relatively low to save weight while vaporizing all the propane in the nozzle. Part of the RP-1 is bypassed in order to minimize channel height and combustion chamber weight while avoiding high coolant jacket pressure drops.

Dump/Film Cooled Chamber

Ground rules for the dump/film cooled chamber designs are listed in Table 6. The dump/film cooled chambers include a short INCO liner which is tightly fitted into the columbium shell. The length of the liner depends on the maximum allowable bulk temperature which is the same as that specified for the regeneratively cooled chambers. Film coolant requirements were based on a chamber design which has the capability of operating safely at mixture ratios within $\pm$ 12 percent of the nominal values, i.e., the amount of film coolant is that which is required to maintain the columbium nozzle temperature at 2400F or less. The combustion chamber is wrapped

with insulation to maintain a maximum temperature of 600F on the outer wall of the insulation. Insulation is applied to an area ratio of 3 at which point the nozzle is radiation cooled with assistance from the film coolant injected by the liner. As with the regeneratively cooled designs, the coated columbium nozzle extends only to an area ratio where the temperature is less than 1600F at which point a titanium nozzle is used to save weight.

Similar to the regeneratively cooled chambers, optimizations of combustion chamber length and film coolant flowrate were performed to optimize performance consistant with temperature and life considerations. The area ratio at which the transition from columbium to titanium takes place in the dump/film chambers is shown in Fig. 10.

ENVELOPE PARAMETRIC DATA

Envelope parameters evaluated are depicted in Fig. 11 with the significant influence parameters provided in Table 7 . Engine length was calculated in terms of a nozzle length, a combustion chamber length and a length which includes the injector thickness and the distance which the oxidizer duct extends above the injector. Both static and dynamic (for gimbaled engine) maximum diameters were calculated. A gimbal angle of 7 degrees was used for calculation of dynamic diameter. The static chamber diameter is twice the radius described by the maximum distance of the ducting from the axis of the chamber. The dynamic chamber diameter closely approximates the dimension for engine installation calculations in the vehicle.

Figure 12 to 15 provide nozzle length data (70% throat to nozzle exit) and Fig. 16 to 19 overall engine length data as a function of thrust, chamber pressure, and expansion ratio. In the engine length data the nominal distance from the combustor throat to the injector was optimized by balancing the effects of injector performance

and film coolant performance degradation. The general effects of propellant combination and cooling method on the combustor length are shown in Table .

The optimizations were conducted for a fixed nozzle length and a 70% bell contour. The data shown in Fig.20a and 20b illustrate some interesting aspects of optimizations for fixed engine length conditions. In Fig.20a results of a reoptimization of combustor length for a fixed engine length of 68 inches are shown. The optimizing parameter is the total program cost based on the factors shown in Table 9 . The results indicate that the optimization made at fixed engine lengths differs very little from the optimization made without consideration of total engine length. The data shown in Fig.20b indicate the effect of nozzle contours on delivered specific impulse. Engine weight considerations may result in flattening this curve somewhat but the light radiation-cooled nozzle weight changes are small. The implication is that for a fixed engine length requirement the combustion chamber may be first optimized without regard for the engine length and the nozzle length should be chosen to completely fill the engine length envelope.

Static and dynamic exit diameters are shown in Fig. 21 through 24 for thrust levels of 3500 through 10,000 pounds. The effects on the dynamic diameter of increasing the nozzle length and of a head-end gimbal are shown in Fig. 25

The maximum diameter of the combustion chamber region is shown in Fig.26 and 27. This diameter is a static diameter. The dynamic diameter is not much larger because of the close proximity to the gimbal point. The diameter of the regeneratively cooled engine is greater than that of the film cooled engine because of the propellant valve orientation which was selected to minimize ducting weight.

Figure 28 provides the engine throat area employed to establish parametric envelope data as a function of thrust, chamber pressure, and expansion ratio.

WEIGHT PARAMETRIC DATA

Weight parameters evaluated are listed in Table 10 with the significant influence parameters provided in Table 11 . Detailed weight calculations were made at design points for each cooling method and the resulting weights scaled parametrically.

Weight data for the throat gimbal and propellant valves are presented in Figs. 29 and 30. The weights of regeneratively cooled thrust chambers NTO/MMH and NTO/50-50 are shown in Fig. 31 . These weights include the injector, combustion chamber and nozzle. The thrust chamber assembly weight increments for LOX/RP-1, LOX/MMH and LOX/N_2H_4 are shown in Fig. 32 and are functions only of thrust and chamber pressure.

Regeneratively cooled engine weight data are presented in Fig. 33 amd 34. The effect of changing nozzle percent length at various area ratios and thrust chamber pressure ratios is shown in Fig. 34 . Engine assembly weights for dump/film cooled thrust chambers are shown in Fig. 36 to 38 for NTO/MMH and LOX/MMH, and LOX/50-50 engine weights are sensitive to mixture ratio over the ranges specified. The engine weight savings which would be effected by eliminating the gimbaling requirements are shown in Fig. 39 At low chamber pressures the gimbal weight of a throat ring gimbal becomes significant and the use of a head-end gimbal should be considered. An engine weight comparison for nominal operating conditions is shown in Fig. 40

INLET PRESSURE PARAMETRIC DATA

The pressure drop components for the fuel and oxidizer sides of regenerative cooled and dump/film cooled engines are shown in Fig. 41 . Feed system pressure drops were patterned after the pressure drops of the LM Ascent Engine. A low-pressure-drop quad-redundant valve is used so that failure of one side of the valve will not result in significant variations in mixture ratio. A minimum injector

orifice pressure drop of 15 percent of chamber pressure at off-design conditions was selected to avoid feed system coupled cooled instability. The performance criteria for evaluating injector pressure drop requirements are listed on Table 12 .

The pressure drop of all components, excluding the injector and coolant jacket is 1.05 x chamber pressure (throat stagnation correction pressure for 2:1 contraction ratio engine) plus 19 psi on the oxidizer side and 15 psi on the fuel side, Fig. 41). A summary of the design conditions and resulting pressure drops in the regenerative cooling jackets is presented in Table 13 . This same data is presented graphically in Fig. 42 Injector orifice pressure drops are shown in Fig. 43. The pressure drops shown are valid for the higher mixture ratios used with O_2/RP-1 and O_2/C_3H_8 because the pressure drops of these injectors do not vary with design mixture ratio. The resultant engine inlet pressure as a function of chamber pressure is presented in Fig. 44 . The variation of inlet pressure with mixture ratio is shown in Fig. 45 for those propellant combinations affected.

PERFORMANCE PARAMETRIC DATA

Performance was determined using JANNAF procedures. Table 14 summarizes the approach used in deriving engine delivered performance. All performance calculations started with a one-dimensional kinetic analysis for the specific propellant combination, chamber pressure, mixture ratio, nozzle area ratio, and thrust level. Boundary layer (drag and heat transfer) and divergence losses in the combustion chamber and nozzle were determined. Propellant vaporization losses were calculated for each of the propellant combinations as functions of chamber lengths. Based upon empirical data, a core mixing efficiency of 0.986 was used for all analyses. This value is reasonable for all propellant combinations through appropriate sizing and spacing of injector elements. The losses resulting from mixture ratio stratification due to film coolant were analyzed as follows: 1) the film coolant flowrate was calculated based on heat transfer analysis;

2) this flowrate was then used to compute a linearly varying mixture ratio profile from a zero value at the wall to the core mixture ratio; 3) a stream tube analysis was then used to calculate the specific impulse versus distance from the wall and to determine an integrated total specific impulse for the chamber. These calculations assumed a 100% cooling effectiveness.

Theoretical one-dimensional equilibrium performance data are presented in Fig. 46 through 48 for reference purposes only. As previously stated performance calculations are based on ODK performance data which is shown in Fig. 49 through 51 . At very low mixture ratios, such as those associated with the analysis of the film coolant boundary layers, ODK performance computations are inaccurate for propellant combinations containing carbon when the quantity of carbon in the exhaust becomes significant. One-dimensional frozen performance data is a better representation of performance under these conditions. The combined ODF and ODK performance data are shown in Fig. 52 with ODF at the lower mixture ratios and ODK at the higher mixture ratios. ODK performance data are plotted in Fig. 53 through 58 with the nozzle expansion area ratio as the abscissa. The effects of thrust level on kinetic performance are indicated on each of these graphs and are shown graphically in Fig. 58.

The effect of film coolant flow on ODK performance is shown in Fig. 58 through 61 for the various propellant combinations. Film coolant requirements are considerably higher for dump/film cooled chambers (Fig. 62) than for supplementing regen cooling. The implications on delivered performance may be seen by noting the effects on the stratification loss curves shown earlier. An additional loss occurs because the dump/film cooled chambers are shorter, resulting in decreased injector efficiency as shown in the vaporization efficiency plots on Fig. 63 . Vaporization efficiency for LOX/RP-1 could not be determined due to computer program inability to model the RP-1 combustion process. Based on empirical data a 20 inch combustor was used on all LOX/RP-1 engines to provide a $\eta_{VAP} \sim 97.5$ percent.

Nozzle geometric and drag losses are shown in Fig. 64 and 65 . The previously shown ODK performance data and losses were combined to yield the overall performance data tabulated in Tables 15 to 20 for the regeneratively cooled systems and in Tables 21 to 23 for the dump/film cooled systems. Delivered specific impulse data for the regeneratively cooled engines are plotted in Fig. 65 to 70, and for the dump/film cooled engines in Fig. 71 to 73 . The effects of thrust level on performance is shown in Fig. 74 and 75 . Performance is degraded at the 3500 pound thrust level because of thehigher film coolant requirements. Representative performance data for constant envelope systems are shown in Fig. 76 to 78 . Although the data shown in Fig. 77 and 78 indicate that high pressure yields high engine performance, it is evident in Fig. 79 that vehicle weight effects result in an optimum P_c value in the 100 to 150 psi range.

SENSITIVITY DATA

The OME configurations were evaluated relative to the effect of propellant inlet pressure and inlet temperature. This was done in two steps. Initially a feasible range of pressure and temperature had to be established based on which the engine would be designed. This range includes engine calibration,nominal tank pressure variation, nominal propellant temperature variations, effect of regulator malfunction, and effect of propellant valve malfunction. From these, engine mixture ratio and chamber pressure ranges are established. For the ranges established the engines are designed to meet engine life requirements (i.e. for the purpose of parametric data dump, mixture ratio ± 12% of nominal,chamber pressure ± 10% of nominal,and maximum inlet temperature of 90F were assumed).

Once the point design is established sensitivity plots were generated of engine mixture ratio and chamber pressure versus inlet pressure and temperature, and delivered specific impulse versus the resultant mixture ratio (insensitive to chamber pressure area range of interest), and engine temperature.

SENSITIVITY TO SELECTION OF OPERATING RANGE

An evaluation of mixture ratio and chamber pressure variations in a regenerative cooled NTO/MMH engine was performed, the results of which are summarized in Table 24 . Engine calibration limits were based on LMA. Inlet pressure variation of ± 4 psia assume seperate control of the NTO and MMH tanks and allow for some temperature differential between tanks along with 1% regulator repeatability. As can be seen, mixture ratios as high as 1.8 o/f result just from normal pressure and temperature excursions. Should an oxidizer side regulator malfunction occur (fail open) which causes the upstream regulator to control pressure (set for 10 psia above downstream regulator in present Space Division

system) , the maximum mixture ratio rises to 2 o/f. With this type malfunction, the coolant flow is not reduced so engine cooling is not a problem. The chamber pressure increases to only 133 psia. Minimum chamber pressure occurs when a valve malfunction closes one of the bipropellant flow paths, inlet pressure is a minimum, and inlet temperature is a maximum. The resultant chamber pressure is 116.6 psia.

The performance penalty associated with designing the OME for anticipated mixture ratio, chamber pressure, and inlet temperature ranges was evaluated.

The restraint imposed on fuel injection temperature (i.e., at least 50F below the bulk boiling temperature for worst-case operation) requires that any increase in the assumed fuel inlet temperature be accompanied by a corresponding decrease in the cooling jacket ΔT. The impact of this consideration is that as design inlet temperature increases, film coolant flow must be increased and/or combustion chamber length must be decreased. Since either of these alternatives is detrimental to delivered specific impulse, an analysis of the effect of fuel inlet temperature on OME performance was conducted for NTO/50-50 and NTO/MMH at nominal operating conditions.

Results of the analysis for the OME thrust chambers are summarized in Fig. 80 . It is apparent in the data for regeneratively cooled chambers that MMH is less sensitive than 50-50 to increased inlet temperature. This result stems primarily from the 20F (and corresponding film coolant flow requirement) advantage MMH has over 50-50 with respect to boiling point at the nominal injection pressure. Thus, although NTO/50-50 is theoretically higher performing than NTO/MMH, the difference in boiling points yields nearly equal performance for a fuel inlet temperature of 90F and a 1.7 second advantage for NTO/MMH for an inlet temperature of 120F. Note that because of the 2 percent minimum film coolant flow constraint, the NTO/MMH system is unaffected by inlet temperatures up to 102F.

For dump/film cooled systems, a similar result is obtained. In these cases, higher inlet temperature requires either that a shorter sleeve (and correspondingly shorter injector-to-throat distance) be utilized at constant coolant flowrate or that a higher coolant flowrate through the same sleeve be used (with a slight increase in sleeve exit-to-throat distance). For either approach, the delivered specific impulse in an NTO/MMH chamber (6K, 125 psia) is reduced approximately one second when fuel inlet temperature is increased from 90F to 120F. Thus, the impact of fuel inlet temperature on specific impulse is compatable to the effect observed in the regeneratively cooled chambers.

Incorporation of off-nominal mixture ratio and chamber pressure operating capability requires that film coolant flows be established for worst-case situations. As a result, the flows are higher than required at nominal conditions.

The maximum penalties paid at nominal conditions for off design capability are shown in Fig. 81 and 82 . As an example, Fig. 81 shows that with a ± 12% mixture ratio and ± 10% chamber pressure variation there is a 1.8 second specific impulse penalty at the nominal design point with NTO/50-50 in the regeneratively cooled chamber (2.1 second loss with off-design capabilities, 0.3 second loss with no capabilities).

NTO/MMH is less sensitive to these mixture ratios and chamber pressure variations (~ 0.3 seconds in regenerative cooled chamber). For the same flexibility a film cooled NTO/MMH engine incurs a 1.0 second performance loss (Fig 82).

SENSITIVITY TO ENGINE INLET CONDITIONS

The sensitivity of mixture ratio and chamber pressure to changes in tank pressure is shown in Fig. 83 to 86 for the nominal OME configurations. The two cooling methods indicate nearly equal sensitivity. In terms of specific impulse, the impact of chamber pressure variation is

negligible; mixture ratio excursions within the expected range may, in contrast, vary specific impulse by several seconds. This effect is shown in Fig. 80

As shown in Fig. 87 , an important aspect of sensitivity to pressure changes is the position of the design mixture ratio with respect to the value for peak performance. This is clearly evident for NTO/50-50, in which a nominal mixture ratio of 1.6 results in oppositely directed sensitivities for the regen/film and dump/film cooled chambers (i.e., a small increase in mixture ratio will improve performance for the regen/film design but hurt performance in the dump/film design). The effects of inlet pressure variations on performance and thrust level are shown in Table 24 . When both tank pressures vary the same amount (as would be expected with a common pressurization system) performance is minimaly affected. The performance of the O_2/MMH film cooled chamber was most sensitive and the NTO/MMH film cooled engine was least sensitive. The sensitivity of thrust to tank pressures is similar for all configurations. Sensitivity of chamber pressure and mixture ratio to propellant temperatures are shown in Fig.88-90. Variations resulting from temperature changes in the ranges shown are seen to be small compared to those resulting from anticipated pressure variations.

Typical examples of wall temperature sensitivity are shown in Fig. 91 and 92 . For the regeneratively cooled chambers, Fig. 91 illustrates that within the expected range of P_c variations, the change in wall temperature at the radiation nozzle attach point is small. Figure 92 indicates the sensitivity of the film cooled chamber to film coolant flowrate . The sensitivity of the regeneratively cooled chamber wall temperature to film coolant flow is much lower (10F for a 20% reduction). Reduction of the primary, regenerative, coolant flow by 20% results in a decrease in the cooling safety factor from 2.0 to 1.6.

TECHNOLOGICAL AND OPERATIONAL FACTORS

In addition to the quantitative parametric data previously shown, the chamber cooling methods and propellant combinations were compared qualitatively with respect to complexity, reliability, maintainability, safety, development risk, and ecological factors.

COMPLEXITY

The regeneratively cooled chamber was rated as slightly more complex than the film cooled design because of the greater number of joints and manifolds and the full length double wall construction. Complexity features associated with the film cooled chamber are tighter tolerances on small channel dimensions, insulation requirements, and potential requirements for propellant sequencing at cutoff to control thermal soakback.

NTO/MMH was estimated to involve the least complexity followed closely by NTO/50-50. The high freezing point of 50-50. The high freezing of 50-50 compa red to MMH and resulting possible requirements for heaters is responsible for its lower rating. Propellant combinations using LOX as the oxidizer are significantly more complex. The primary factors increasing the complexity associated with LOX engines are the requirements for ignition and purging systems, and the complexities associated with c ryogenic propellant operation, i.e., seals, insulation to reduce LOX boiloff, and configurations to prevent fuel freezing. Handling RP-1 would be less complicated than the amines becuase of its lower vapor pressure and toxicity.

RELIABILITY

Mission reliability includes the probability of accomplishing the mission by avoiding component or system failures and by meeting specified performance goals. A detailed reliability assessment of the regenerative and film cooled engines is provided in Table 26 . The regeneratively cooled engine was estimated to be more reliable primarily because of the redundant nature of the double wall, lower operating temperatures, the significantly greater development and operational experience, the lower sensitivity to contamination, and the higher performance. Simplicity is the primary reliability asset of the film cooled engine.

A detailed reliability assessment of the candidate propellant combinations is provided in Table 27 . The NTO/MMH and NTO/50-50 propellant combinations were determined more reliable than the propellants using LOX for the same reasons described for the complexity evaluation. In addition, the tendency of LOX propellant combinations to produce detonable gels with RP-1 and amine fuels reduces system reliability. The higher performance of the propellants using LOX is a favorable reliability factor. Development and operational experience with LOX/RP-1 is significant but many of the OME operating parameters and requirements are outside of the areas in which the experience is applicable.

MAINTAINABILITY

Maintainability comparisons were based on lowest maintenance and servicing (including propellants) requirements and ease of performing maintenance if required. Detailed comparisons are shown in Table 28 (compares cooling concepts) and Table 29 (compares propellant combinations). Maintainability of the regenerative and film cooled engines are approximately equal. Both require no scheduled maintenance. The regeneratively cooled chamber, with higher reliability, will require less unscheduled maintenance,

and replacement of the bolted nozzle on the regeneratively cooled chamber is easier than the welded nozzle on the film cooled chamber. More care must be taken in inspecting the film cooled chamber post flight to avoid damaging the coated columbium in the high heat flux combustor region. Combustor replacement is more difficult in the regenerative cooled engine, however, if required.

NTO/MMH and NTO/50-50 have approximately equal reliabilities, no scheduled maintenance, and equal ease of performing unscheduled maintenance. The greater complexity of the system using NTO/50-50 propellants was offset by the lower propellant cost of 50-50. The lower reliability of the propellants which use LOX lowers the ranking of the LOX/RP-1 and LOX/Amine fuel systems. The cost and lower toxicity level of RP-1 relative to the amine fuels make it more maintainable in a LOX system.

SAFETY

Handling, testing, and operating aspects of safety were evaluated for the candidate cooling modes (Table 30) and propellants (Table 31). The double wall features of the regenerative cooled chamber provides a safety margin, should a hot wall crack occur, not present in the film cooled engine which is more susceptible to this failure mode due to the high operating temperatures and requirement for a coating for material compatibility.

Reliability considerations indicate a greater safety for NTO/MMH and NTO/50-50. Although LOX is not toxic it does form detonable gels, is flammable with fabrics, and contact with the skin is harmful. The low toxicity and vapor pressure of RP-1 is a safety advantage relative to amine fuels.

DEVELOPMENT RISK

The relative certainty of providing an OME which will meet the specified requirements with a modest development program was evaluated based on existing technology and anticipated development requirements. Analytical procedures have been developed for both regenerative and film cooled engines. Both cooling methods require demonstration at OME conditions. However, considerably more experimental data are available on technology, developmental, and operational regeneratively cooled engines compared to buried film cooled engines. Accordingly, the regeneratively cooled engine appears to have the lower development risk.

Critical areas in OME development with NTO/Amine propellants are:

1. Propellant valve cycle life and reusability in the Space Shuttle environment
2. Stability (the ability to maintain a stable system without baffles will significantly reduce system complexity and insure life potential).
3. Restart capability (NTO or 50-50 freezing during shutdown would restrain restart time periods)
4. Columbium coating compatibility with the shuttle duty cycle and environment (disilicide coating durability and cyclic capability demonstrated in laboratory tests; however, without intermittent exposure to launch and recovery environment, Ref. 4).

When comparing propellant combinations,NTO/MMH and NTO/50-50 have the lowest development risk because of the large available technology base. LOX/RP-1, although widely used in larger non-reusable,non-restartable

engines presents more areas where technologies applicable to the OME condition have not been adequately explored. LOX/MMH and LOX/50-50 have the smallest technology base no operational experience-and hence, the highest development risk.

Critical technology areas associated with LOX/RP-1 and LOX/Amine systems are:

1. Propellant valve cycle life and reusability in the Space Shuttle environment; isolation of propellants to preclude freezing of the fuel in a mechanically linked configuration.
2. Stability criteria not established for OME size engine.
3. Restart capability (purging probably required to preclude formation of gels and reduce soakback to LOX if rapid restart desired).
4. Columbium coating compatibility with the shuttle duty cycle and environment.
5. Ignition
6. LOX/RP-1 performance and heat flux (carbon deposition) characteristics at OME conditions.

ECOLOGY

A complete comparative analysis of the ecological impact of various propellant combinations would include the following aspects: 1) impact of production of the fuels and oxidizers on the earth environment; 2) impact of the use of the various propellant combinations on the earth environment during ground testing; and 3) impact of the use of the propellant combinations on the space environments during flight test and operation.

Evaluation of the impact of production of the propellants on the earth environment is beyond the scope of the present study. Such an evaluation would include not only analysis of the byproducts of production of the

propellants themselves but also the ecological impact of obtaining or manufacturing all the chemicals from which the propellants are produced, tracing the processes back to the natural sources of all the constituents. Energy consumption and environmental implications would also have to be considered in the evaluation.

The quantity of propellants and combustion products exhausted into space by the OME is so small and distributed as to constitute a negligible ecological factor. Thus, only the impact of using the various propellants on the earth environment during ground test was evaluated.

The combustion products of the various propellant combinations were determined for expansion to a nozzle exit area ratio corresponding to a 50 inch diameter and a 6K thrust level. Propellant combinations using NTO as the oxidizer were evaluated at a chamber pressure of 125 psia and mixture ratios corresponding to equal volume propellant tanks. Propellant combinations using LOX as the oxidizer were evaluated at a chamber pressure of 100 psia and mixture ratio corresponding to the optimum performance level. The compositions are calculated on the basis of one-dimensional kinetic analyses. The temperature of the combustion gases at these exit conditions is less than 1,000F so that further reactions of these constituents between themselves do not take place. However, afterburning of combustible products such as H_2 and CO will probably occur.

The composition of the combustion products of the various propellant combinations are shown in Table 32 . The two constituents listed in this table which directly affect air quality are carbon monoxide (CO) and nitrogen oxide (NO). Because the engines are run relatively fuel rich compared to diesel or gasoline fueled power plants, the

percentages of nitrogen oxide in the exhaust products are quite low and the percentages of carbon monoxide are relatively high. However, it is expected that afterburning of the carbon monoxide will occur to the extent that all or most of the carbon monoxide will be converted to carbon dioxide. The OME at the 6K thrust level produces the equivalent of approximately 1.2×10^5 horsepower. Burning approximately 20 lbs of propellant per second the propellant combination which produces the maximum amount of NO, LOX/50-50, emits approximately 5×10^{-5} grams of NO per hour. The resultant emission rate of 3.3 grams per equivalent horsepower hour is less than the 1975 emission standard of 5 grams per horsepower hour applied to heavy duty vehicles.

Approximately 870,000 pounds of NTO/MMH would be used during the OME development and qualification program which would result in producing 3500 pounds of NO during a period of 3 years. The total propellants expended in space by the OME during 11 years of operation (400 flights) is approximately 10^7 pounds which results in 40,000 pounds of NO in space. The present motor vehicle emission rate into the Southern California basin of NO is nearly 80,000 pounds/hour. At the nozzle exit the carbon monoxide content of LOX/RP-1is approximately 4 times the 1975 standard of 25 per grams per horsepower hour. However, it is anticipated that most of this carbon monoxide would burn off and the resultant concentration be within the standard value.

Therefore, it appears that the emissions of all the candidate propellant combinations will be acceptable by the 1975 heavy duty vehicle standards. However, for purposes of comparison, the propellant combinations using oxygen produced more NO than the propellant combinations using NTO and the propellant combinations with MMH. LOX/50-50 produces approximately 3 times as much NO as the NTO/MMH propellant combination. With respect to CO, the oxidizer appears to be less than significant than the fuel. 50-50 results in slightly less CO than MMH. LOX/RP-1 produces approximately

twice as much CO as NTO/MMH. Although not shown in the Table, which is for 100 percent injector efficiency, actual injectors produce small amounts of finely divided unburned carbon with LOX/RP-1 which results in sooty exhaust.

COMPARISON

The comparison of regenerative and film cooled engines based on technological and operational factors is summarized in Table 33. Regenerative cooling was judged better considering all factors. The same considerations led to selection of NTO/MMH as the best propellant combination followed closely by NTO/50-50. The order of preference for the remaining candidates was O_2/RP-1, O_2/MMH, and O_2/50-50. The last two propellant combinations were closely ranked.

PROGRAM COMPARISON

Program comparison for the candidate propellants and thrust chambers were made by establishing a baseline program and determining the differences in program cost, program schedule and operational costs for the alternates. The regeneratively cooled N_2O_4 Orbit Maneuvering Engine was selected as the baseline system.

The development program is divided into design feasibility test (DFT), design verification test (DVT), and qualification test phases. The development status of the engine components coupled with the propellant combination determined the level of design feasibility testing required to proceed to DVT and qualification. The number of engines submitted to DVT and qualification is the same for all programs. Additional components, however, were submitted to DVT or qualification testing depending on the development status of the component and the associated development risk. The following assumptions were used to establish the development program for each propellant-thrust chamber combination:

1. The test programs are success oriented.
2. Current SS/OME Reusable Thrust Chamber technology programs will demonstrate the feasibility of the regeneratively and film cooled thrust chambers and injectors with NTO/MMH.
3. The thrust level of the OME will be different than the 6000 pounds level being tested in the SS/OME Reusable Thrust Chamber programs.
4. An electrical ignition system such as an augmented spark igniter (ASI) will be required for non-hypergolic propellants to provide multiple restart capability.
5. There will be no significant ignition development for LOX/Amine or LOX/RP-1 prior to program go-ahead.
6. Multiple restart capability has not been established for LOX/Amine or LOX/RP-1.

7. Preliminary LOX/MMH combustion performance data as a function of chamber length and L* will be available. The feasibility of the regeneratively cooled thrust chamber and injector with LOX/MMH will have been demonstrated. (This is predicated on the assumption that some technology, either propulsion company in-house or NASA funded, would be required before a commitment to these propellants would be made for the Space Shuttle application).
8. Combustion performance and heat flux with carbon deposition as a function of chamber length is not available for LOX/RP-1.
9. Data obtained with N_2O_4/MMH is directly applicable to N_2O_4/50-50.
10. The feasibility of the long-life ball and seat configuration will have been demonstrated for the propellant valve prior to program go-ahead.

DEVELOPMENT PROGRAM

The development program schedules for the various thrust chambers and propellant combinations are shown in Fig. 93 through 96 . The same development program is shown (Fig. 93), for the regeneratively cooled thrust chamber and injector and the film cooled thrust chamber and injector with N_2O_4/50-50. The demonstrated feasibility and hardware fabrication flow times are approximately the same; consequently the major differences between the programs will be hardware cost and propellant cost.

The programs for the non-hypergolic propellants (Fig. 94 to 96)all require the development of an ignition system to provide multiple restart capability. For the purposes of scheduling and cost estimates, it was assumed that an augmented spark igniter (ASI) would be used. The ASI has been developed and qualified for oxygen and hydrogen, and some work has been done with LOX/RP-1; however, no work has been done with LOX/amines. The schedules reflect a thorough DFT program for the ignition system since development of the start and cutoff sequence are critical where a restart is required. RP-1, MMH, and 50-50 contain

carbon and, when combined with LOX, can form an explosive gel. Residual propellants in the thrust chamber-injector during restarts could result in a hazardous condition. Purge sequencing for both the ASI and the injector thrust chamber will be part of the ignition system. A more conventional ignition system (hypergolic) will be used to start injector testing.

Injector and regeneratively cooled thrust chamber feasibility will have been demonstrated for LOX/MMH. The similarity between MMH and 50-50 and the significant injector performance experience with these fuels would indicate comparable program schedules and hardware cost, with the major difference being propellant costs. The LOX/MMH film cooled engine will utilize the injector performance data from the regeneratively cooled thrust chamber feasibility testing; however, additional testing will be required to demonstrate compatibility and overall performance of the injector-film cooled thrust chamber combination. Release of the initial flight thrust chamber design for each thrust chamber type will be delayed until the optimum combustion zone configuration can be established.

Basic combustion zone length and L* data are required for the LOX/RP-1 propellant combination before the flight thrust chamber design can be released. This performance optimization and the special testing to evaluate purge effectiveness with the low vapor pressure RP-1 will result in the longest program through qualification. The propellant costs are considerably reduced for the LOX/RP-1 propellant combination.

Propellant costs are not normally considered as part of a program cost since the propellants are government furnished, and the propellant combination is usually specified. The wide variety of propellants being considered for the OME,however, could make program propellant costs an important factor. Table 34 summarizes the propellant costs for the OME program for each propellant combination. This table is based on the propellant quantities supplied in support of the NTO/MMH OME proposal to a vehicle contractor. These costs are based on the August 1972 government supplied price list except MMH which is based on production levels commensurate with the development and flight usage schedules.

MAINTENANCE

A preliminary maintenance analysis for the OME during the Space Shuttle program operational phase was performed (Ref. 5). This analysis determined the approximate OME maintenance cost per orbiter vehicle mission duty cycle using N_2O_4 and MMH with a regenerative or film-cooled thrust chamber. All components other than the engine assembly (propellant tanks, regulators, electrical controllers, etc.) were considered as vehicle components. This study was expanded to include the alternate propellant combinations and the more complex engine assembly required for the non-hypergolic propellants. The maintenance costs established because of these new requirements were handled as incremental costs over the baseline study.

Routine and corrective maintenance tasks were investigated as a part of the study, and labor and material costs to support each task were estimated. The routine maintenance tasks, to be performed on a scheduled basis after each flight, were identified by evaluating OME design and vehicle operational concepts. The corrective maintenance tasks, to be performed on an unscheduled basis when an OME problem occurs, and task frequency were identified from OME problem and problem rate projections. The corrective maintenance tasks that were identified consisted of: (1) those required in the vehicle to return the engine to a serviceable condition; and (2) those required in the site engine shop and the factory to restore repairable OME hardware to a serviceable condition and return it to the site, or if not repairable, to provide serviceable replacement hardware. The man-hours and material costs required to perform each of the routine and corrective tasks were then estimated and summed. The details of these maintenance studies are included in Table 35 to 39 . The maintenance cost summary for one orbiter mission cycle is shown in Table 40 . Because of the infrequency of the in-vehicle and site engine shop maintenance tasks, a factor of 1.5 is used to cover the less tangible aspects of cost. This same factor is used for routine maintenance.

LOGISTICS

The logistics support for the OME falls into the following categories; technical manuals, spares management, logistics engineering and technical services, training, and field engineering support. The increased complexity of the OME when using non-hypergolic propellants will increase the logistics cost of the technical manuals, spares management and training by about $19,000. It is suggested that a full-time field engineer be assigned to the vehicle contractor facility during initial installation and checkout, and would subsequently be relocated at the launch site to support the operational program. The increased complexity of the non-hypergolic OME will not require additional field engineering.

PROGRAM COST AND SCHEDULE COMPARISON SUMMARY

Table 41 is a summary of the program cost and schedule comparisons for the propellant combinations and thrust chambers being considered for the OME. For this comparison, the N_2O_4/MMH propellant combination with the regeneratively cooled thrust chamber is considered the baseline program. All other program schedules, program costs and operational costs are incremental values from the baseline. The program costs for manpower and hardware through qualification are listed separately from the propellant costs since propellants are normally government furnished.

As expected the program with the least demonstrated feasibility testing, the LOX/RP-1 propellant combination with a regeneratively cooled thrust chamber, resulted in the greatest increase in cost. The other development programs with non-hypergolic propellants also cost $1 mission more than the baseline.

The N_2O_4/MMH film cooled thrust chamber was the least costly program, slightly less than the baseline N_2O_4/MMH regeneratively cooled thrust chamber program, due to lower thrust chamber costs. When the propellant costs for the development program are considered, the N_2O_4/50-50 programs are the lowest in cost and the LOX/MMH programs are the highest (by 1.7 to 1.8 million dollars).

Table 42 presents an estimated total program cost including propellants through the first 100 flights. This table is presented primarily to show the influence of propellant cost on the overall program. The lowest cost program is the N_2O_4/50-50 propellant combination with either type of thrust chamber. The costliest programs are the LOX/MMH programs due to the added cost of the development and the high price of the MMH fuel. LOX/RP-1 program cost is $600,000 less than the baseline program due entirely to the low cost of propellants. As previously stated, this program cost and schedule comparison is based on a successful solution of all the problems associated with each propellant combination; consequently the impact of the program costs may be conservative. One factor that has to be considered with this comparison is the development risk associated with each propellant and thrust chamber combination, especially the non-hypergolic propellant combinations.

CONCLUSIONS AND RECOMMENDATIONS

The selection of the optimum cooling mode and propellant combination for the OME is effectively performed in 4 steps:

1. Selection of optimum amine fuel for use with NTO or LOX oxidizers.
2. Selection of optimum cooling mode in NTO and LOX systems
3. Selection of "best" alternate propellant combination to the baseline NTO/amine fuel system.
4. Comparison of the baseline and alternate propellant combination leading to the selected OME configuration.

MMH was chosen as the optimum amine fuel due to its lower freezing point, better cooling capability, lower sensitivity to engine operating conditions, and better thermal stability. In a regenerative cooled system MMH performance is comparable to 50-50 whereas in a film cooled system it is superior. The only advantage of 50-50 is lower propellant cost. The present cost difference will be reduced from $1.75/lb to about $0.50/lb if MMH is selected for the Space Shuttle vehicle.

Regenerative cooling was selected over film cooling due to higher performance, greater reliability, increased safety, and a lower development risk. The film cooled systems primary advantage is that of simplicity but the resultant cost savings during development and operation is negligible.

Of the alernate propellant combinations LOX/RP-1 was superior relative to technology status and engine operation features. The LOX/MMH system provided minimum system weight. The baseline NTO/MMH system, however, provides minimum development risk, is the least complex, the most reliable, and safest system. In addition it is lighter than the LOX/RP-1 system.

Based on these studies, therefore, a regenerative cooled NTO/MMH engine is recommended for the Space Shuttle Orbit Maneurvering System.

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3. Progress Report Number 1, Space Shuttle Orbit Maneuvering System Trade Studies, McDonnell Douglas Astronautics Company-East, St. Louis, Missouri, August 1972.

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TABLE 1. DATA DUMP CONFIGURATION

PROPELLANTS	F, K	Pc, PSIA	O/F	ε	COOLING
NTO/MMH	6 (36 cases)	100,125,150,200	1.65,1.45,1.85	40,80,120	REGEN, FILM
	3.5 (3 cases)	125	1.65,1.45,1.85	80	
	8 (3 cases)	125	1.65,1.45,1.85	80	
	10 (3 cases)	125	1.65,1.45,1.85	80	
NTO/50-50	6 (36 cases)	100,125,150,200	1.6,1.4,1.8	40,80,120	REGEN, FILM*
	3.5 (3 cases)	125	1.6,1.4,1.8	80	
	8 (3 cases)	125	1.6,1.4,1.8	80	
	10 (3 cases)	125	1.6,1.4,1.8	80	
LOX/MMH	6 (36 cases)	75,100,125,200	1.2,1.0,1.4	40,80,120	REGEN, FILM
	3.5 (3 cases)	100		80	
	8 (3 cases)	100		80	
	10 (3 cases)	100		80	
LOX/50-50*	6	100 (3 cases)	1.2,1.0,1.4	80	REGEN
	6	75	1.2	80	
	6	125	1.2	80	
	6	200	1.2	80	
LOX/N_2H_4**	6	100(3 cases)	0.7,0.8,1.0	58	REGEN
		75	0.8	43	
		125	0.8	73	
LOX/RP-1	6 (36 cases)	75,100,125,200	2.5,2.3,2.8	40,80,120	REGEN
	3.5 (3 cases)	100	2.5,2.3,2.8	80	
	8 (3 cases)	100	2.5,2.3,2.8	80	
	10 (3 cases)	100	2.5.2.3,2.8	80	
LOX/C_3H_8**	6	100(3 cases)	2.6,2.4,2.8	58	REGEN
		75	2.6	43	
		125	2.6	73	

* abbreviated range
** from preliminary data dump

TABLE 2. OME POINT DESIGN CHARACTERISTICS

Propellant	NTO/50-50	NTO/50-50	NTO/MMH	NTO/MMH	O_2/MMH	O_2/MMH	O_2/C_3H_8	O_2/RP-1	O_2/N_2H_4	O_2/50-50
Cooling Method	R/F/R	D/F/R	R/F/R	D/F/R	R/F/R	D/F/R	R/R	R/R	R/R	R/F/R
Chamber Pressure, psia	125	125	125	125	100	100	100	100	100	100
Expansion Ratio	72	72	72	72	58	58	58	58	58	58
Mixture Ratio	1.6	1.3	1.65	1.65	1.2	1.0	2.6	2.5	0.8	1.2
Delivered I_s, sec	313.9	297.9	313.0	304.8	331.6	318.5	339.3	324.0	331.5	328.3
Engine Weight, lb	185	150	185	150	210	180	258	223	207	210
Engine Length, in.	73	70	73	71	74	72	77	86	77	72
Rad. Attach ϵ	7	3	7	3	6	3	21	5	6	6
Extension Material	Cb/Ti	Cb/Ti	Cb/Ti	Cb/Ti	Cb/Ti	Cb/Ti	Ti	Cb/Ti	Cb/Ti	Cb/Ti
Combustor Length, in.	11	8	11	9	12	10	15	24	15	10
Contraction Ratio	2	2	2	2	2	2	2	2	2	2
Liner Material	CRES	CRES	CRES	CRES	CRES	CRES	Copper	CRES	CRES	CRES
Min. Channel Height, in	.062	.025	.062	.025	.068	.025	.160	.250	.145	.071
Number of Channels	180	514	180	514	202	565	172	202	202	202
Film Coolant Flow, pct	2.9	13.5	2.0	8.4	2.6	9.4	0	0	0	2.7
Sleeve Length, in.	-	2.7	-	2.2	-	2.7	-	-	-	-
Engine Inlet Pressure Ox/Fuel, psia	169/195	169/177	169/189	169/177	138/153	138/138	138/159	138/144	138/150	138/156
Cooling Jacket ΔP, psi	21	6	16	6	14	7	20	3	11	17
Injector Ox ΔP, psia	23	23	23	23	18	18	23	23	18	18
Injector Fuel ΔP, psi	27	30	27	30	18	18	23	23	18	18

R/F/R Regen/Film/Radiation

D/F/R Dump/Film/Radiation

R/R Regen/Radiation

F = 6000 lb

D_{exit} = 50 in.

7 Degree Gimbal

70 Percent Nozzle

TABLE 3. OVERALL OME DESIGN POINT COMPARISON OF COOLING METHODS

PROPELLANTS	NTO/MMH (MR=1.65)		NTO/50-50 (MR=1.60)		O_2/MMH	
COOLING	REGENERATIVE	FILM	REGENERATIVE	FILM	REGENERATIVE (MR=1.2)	FILM (MR=1.0)
I_s, sec	313.1	304.8	313.9	297.9	330.9	318.0
ΔW_{I_s}, lb	Ref.	+681	Ref.	+1312	Ref.	+877
W_{eng}, lb	185	150	185	150	210	180
ΔW_{eng}, lb	Ref.	-70	Ref.	-70	Ref.	-60
Max. P_{in}, psia	189	177	195	177	153	138
$\Delta W_{P_{in}}$, lb	Ref.	-71	Ref.	-106	Ref.	-105
$\Sigma\Delta W$, lb	Ref.	+540	Ref.	+1136	Ref.	+712

OVERALL OME DESIGN POINT COMPARISON OF PROPELLANT COMBINATIONS

PROPELLANTS	NTO/MMH	NTO/50-50	O_2/MMH	O_2/50-50	O_2/RP-1
OMS Weight, lb	27,350	27,300	27,050	27,240	27,790

TABLE 4. GENERAL GROUND RULES

ENGINE ASSEMBLY INCLUDES	THRUST CHAMBER
	INJECTOR (IGNITER FOR LOX)
	THROAT GIMBAL RING (7 DEGREE)
	SERIES/PARALLEL VALVE
	DUCTS
COOLING CONCEPTS	REGENERATIVE AND DUMP/FILM
MINIMUM CYCLE LIFE CAPABILITY	1000 CYCLES
OPERATING DURATION	15 HOURS
CONTRACTION AREA RATIO	2
NOZZLE LENGTH	70 PERCENT
RADIATION SKIRT MATERIAL	COATED COLUMBIUM $1600 < T \leq 2400$ F
	COATED TITANIUM $T \leq 1600$ F
VALVE AND INJECTOR MATERIAL	CRES
VALVE ACTUATION	PNEUMATIC

TABLE 5. REGENERATIVE COOLED CHAMBER GROUND RULES

- $\epsilon_c = 2$
- UPPASS COOLING
- CHANNEL WALL WITH ELECTROFORMED CLOSEOUT
- COPPER LINER FOR LOX/C_3H_8
- CRES LINER FOR ALL OTHER PROPELLANTS
- 0.03 INCH HOT WALL THICKNESS
- CHAMBER LENGTH OPTIMIZED
- AREA RATIO BASED ON RADIATION COOLING
- C_3H_8 AREA RATIO ALSO BASED ON VAPORIZING ALL COOLANT IN NOZZLE ($Q/A < 1$ BTU/IN^2 SEC)
- COLUMBIUM AND TITANIUM ATTACH POINTS

P_c, PSIA	75	100	125	150	200	
ϵ_R	4	6	7	8	11	(COLUMBIUM NOZZLE)
ϵ_R	24	31	31	47	62	(TITANIUM NOZZLE)

- C_3H_8 AND RP-1 USE BYPASS CIRCUIT
- MAXIMUM $T_{BULK} = T_{BOILING} - 50$ F FOR 50-50 AND MMH, 280 F FOR N_2H_4
- SUPPLEMENTARY FILM COOLING FOR NTO/50-50, NTO/MMH AND LOX/MMH
- COOLANT REQUIREMENTS BASED ON VARIATIONS IN P_c OF $\pm$ 10%, MR OF $\pm$ 12%, AND PROP INLET TEMPERATURE OF 40 TO 90 F

TABLE 6. DUMP/FILM COOLED CHAMBER GROUND RULES

- INCO DUCT
- DUCT LENGTH DEPENDS ON MAXIMUM T_{BULK}
- SAME T_{BULK} AS REGEN
- COOLANT REQUIREMENTS BASED ON VARIATIONS IN P_c OF $\pm$ 10%, MR OF $\pm$ 12%, AND INLET TEMPERATURE OF 40 TO 90 F
- CHAMBER LENGTH OPTIMIZED

P_c	75	100	125	150	200
ε_R (NTO/MMH)	10	11	12	13	16
ε_R (LOX/MMH)	9	10	11	12	15

- INSULATED TO LIMIT OUTER WALL TO 600 F TO $\epsilon = 3$
- COLUMBIUM/TITANIUM SHELL AND NOZZLE
- COLUMBIUM/TITANIUM TRANSITION POINT

TABLE 7. *ENVELOPE FUNCTIONS*

- ENGINE LENGTH = L_{NOZZLE} + $L_{CHAMBER}$ + $L_{INJECTOR}$ + $L_{DUCT\ OR\ GIMBAL}$

  $f(F/P_C, \epsilon, \%\ Bell)$ $f(F, P_C, O/F, Prop)$ $f(F)$ $f(F)$

- EXIT DIAMETER
  - STATIC: $D_{ES} = f(F/P_C, \epsilon)$
  - DYNAMIC: $D_{ED} = f(F/P_C, \epsilon, \%\ L,$ GIMBAL ANGLE AND LOCATION$)$

- CHAMBER DIAMETER
  - STATIC: $D_{CS} = f(F/P_C, \epsilon_C)$
  - DYNAMIC: $D_{CD} \approx D_{CS}$

TABLE 8. INFLUENCE OF COOLING METHOD AND PROPELLANTS ON COMBUSTOR LENGTH

F = 6000 LB, NOMINAL P_C

PROPELLANT COMBINATION	COOLING CONCEPT	COMBUSTOR LENGTH, IN.		
		<NOM'L MR	NOM'L MR	>NOM'L MR
NTO/MMH	REGEN	12	11	10
	DUMP/FILM	10	9	8
NTO/50-50	REGEN	12	11	10
	DUMP/FILM	6	6	6
LOX/MMH	REGEN	14	12	10
	DUMP/FILM	10	9	9
LOX/50-50	REGEN	12	10	8
LOX/N_2H_4	REGEN	15	15	15
LOX/RP-1	REGEN	24	24	24
LOX/C_3H_8	REGEN	15	15	15

TABLE 9. **OME TRADE FACTORS**

PARAMETERS	EFFECT OF CHANGING PARAMETER ON	
	PAYLOAD WEIGHT	ORBITER & BOOSTER PROGRAM COST
SPECIFIC IMPULSE, I_S	87 LB/SEC	1,314,000 $/SEC
INLET PRESSURE		
BOTH TANKS, P_2	9.7 LB/PSI	179,000 $/PSI
ONE TANK , P_1	4.8 LB/PSI	90,000 $/PSI
ENGINE WEIGHT, W_E	2.8 LB/LB	48,000 $/LB

PARAMETERS	TRADE FACTORS AT CONSTANT	
	PAYLOAD	ORBITER & BOOSTER COST (USED FOR OPTIMIZATIONS)
P_2/I_S, PSI/SEC	9.0	7.5
P_1/I_S, PSI/SEC	18	15
W_E/I_S, LB/SEC	31	23
W_E/P_2, LB/PSI	-3.5	-3.7
W_E/P_1, LB/PSI	-1.7	-1.8

REF. ΔV = 950 FPS; W_G = 285,000 POUNDS; I_S = 315 SEC;
0.114 LB ORBITER INERT/LB PROPELLANT

TABLE 10. ENGINE WEIGHT

• VALVE	• QUAD-REDUNDANT PROPELLANT VALVES • PNEUMATIC SOLENOIDS • POSITION INDICATORS
• GIMBAL	• THROAT GIMBAL RING • MOUNTING PROVISIONS
• THRUST CHAMBER ASSEMBLY	• COMBUSTION CHAMBER • INJECTOR • MANIFOLDS • IGNITER (FOR LOX)
• ENGINE ASSEMBLY	• ALL ABOVE PLUS • DUCTING • ELECTRICAL AND INSTRUMENTATION WIRING

TABLE 11. WEIGHT FUNCTIONS

- ENGINE WEIGHT $= W_{CHAMBER} + W_{NOZZLE} + W_{INJECTOR} + W_{GIMBAL} + W_{VALVE} + W_{DUCTS} + MISC$

  - $W_{CHAMBER} = f(F, P_C, \epsilon_{RAD}, \epsilon_C, t_{INSUL})$
  - $W_{NOZZLE} = f(F, P_C, \epsilon, \epsilon_{RAD}, \text{PERCENT BELL})$
  - $W_{INJECTOR} = f(F, P_C, \epsilon_C)$
  - $W_{VALVE} = f(F/I_S, o/f)$
  - $W_{GIMBAL} = f(F, P_C, \epsilon_C)$
  - $W_{DUCTS, ETC} = f(F, P_C, \epsilon_C, o/f)$

TABLE 12. INJECTOR PRESSURE DROP CRITERIA

PROPELLANT COMBINATION	INJECTOR PATTERN	STABILITY MINIMUM $\Delta P/P_C$ (ORIFICE)*	INJECTOR DESIGN CRITERIA
NTO/50-50,MMH	UNLIKE DOUBLET	15	MIXING: RUPE OPT VAPORIZATION: NAS7-726
LOX/50-50,MMH	LIKE DOUBLET	15	MIXING: FO4(611)-67-C-0081 VAPORIZATION: NAS7-726
LOX/RP-1	LIKE DOUBLET	15	
LOX/N_2H_4	LIKE DOUBLET	15	
OR			
LOX/C_3H_8	CONCENTRIC TUBE (REGEN) OR LIKE DOUBLET (FILM)	15	NAS9-9528 NAS3-12051

*Occurs only at maximum off-design excursion.

TABLE 13. OME PARAMETRIC REGENERATIVE COOLING DATA

PROPELLANTS	F (LBS)	P_c (PSIA)	O/F	L_{CZ} (IN)	$\dot{W}_{F.C.}$ %	ΔP_j (PSI)	T_{sub}* F
NTO/MMH	6000	125	1.65	11	2.0	15	62
			1.45	12		16	65
			1.85	10		15	59
		100	1.65	11		12	50
			1.45	12		12	54
			1.85	10		12	50
		150	1.65	11		22	71
			1.45	12		22	75
			1.85	10		22	68
		200	1.65	11		33	71
			1.45	12		30	76
			1.85	10		34	68
	8000	125	1.65	12		15	71
			1.45	13		16	76
			1.85	11		15	68
	10000		1.65	13		17	77
			1.45	14		18	81
			1.85	12		16	73
NTO/50-50	6000	125	1.60	11	2.9	21	50
			1.40	12	2.4	22	50
			1.80	10	3.2	21	50

* At most severe off design conditions

TABLE 13. OME PARAMETRIC REGENERATIVE COOLING DATA (Continued)

PROPELLANTS	F (LBS)	P_c (PSIA)	O/F	L_{CZ} (IN)	$\dot{W}_{F.C.}$ %	ΔP_j (PSI)	T_{sub}* F
NTO/50-50	6000	100	1.60	11	3.7	12	50
		150	1.60	11	2.3	28	50
		200	1.60	11	2.3	36	50
O_2/MMH	6000	100	1.2	12	2.6	14	50
			1.0	14	2.7	15	50
			1.4	10	2.2	14	50
		75	1.2	10	2.3	10	50
			1.0	12	2.5	11	50
			1.4	9	2.5	10	50
		125	1.2	12	2.0	23	50
			1.0	15	2.0	23	50
			1.4	11	2.0	24	50
	8000	100	1.2	13	2.0	16	55
			1.0	15	2.0	18	56
			1.4	11	2.0	15	57
	10000		1.2	2.0	14	18	62

* At most severe off design conditions

TABLE 13. OME PARAMETRIC REGENERATIVE COOLING DATA (Continued)

PROPELLANTS	F (LBS)	P_c (PSIA)	O/F	L_{CZ} (IN)	$\dot{W}_{F.C.}$ %	ΔP_j (PSI)	T_{sub}* F
O_2/MMH	6000	100	1.0	16	2.0	21	64
			1.4	12	2.0	16	63
O_2/50-50	6000	100	1.2	12	4.6	14	50
			1.0	14	4.4	15	50
			1.4	10	3.6	14	50
		75	1.2	12	4.6	10	50
		125	1.2	12	4.3	21	50
O_2/N_2H_4		100	0.8	15	0	14	99
			0.7	15	0	13	108
			1.0	15	0	14	80
		75	0.8	15	0	10	111
		125	0.8	15	0	16	88
O_2/C_3H_8		100	2.6			22	
			2.4			22	
			2.8			22	
		75	2.6			19	
		125	2.6			28	

* At most severe off design conditions

TABLE 13. OME PARAMETRIC REGENERATIVE COOLING DATA (Concluded)

PROPELLANTS	F (LBS)	P_c (PSIA)	O/F	L_{CZ} (IN)	$\dot{W}_{F.C.}$ %	ΔP_j (PSI)	T_{sub}* F
O_2/RP-1	6000	100	2.5	24	0	5	
			2.3				
			2.8				
		75	2.5				
			2.3				
			2.8				
		125	2.5			6	
			2.3				
		200	2.5			8	
			2.3				
			2.8				
	8000	100	2.5			7	
			2.3				
			2.8				
	10000		2.5			8	

* At most severe off design conditions

TABLE 14. ENGINE PERFORMANCE

$$ISP_{DEL} = ISP_{ODK} \left(\eta_{ISP_{DIV}} \times \eta_{ISP_{MIX}} \times \eta_{ISP_{VAP}} \right) - \Delta ISP_{BL} - \Delta ISP_{STRAT}$$

ISP_{DEL} = DELIVERED SPECIFIC IMPULSE

ISP_{ODK} = ONE DIMENSIONAL KINETIC SPECIFIC IMPULSE, INCLUDES IMPURITIES AND HEAT ADDITION (REGENERATIVE COOLED ENGINE)

$\eta_{ISP_{DIV}}$ = DIVERGENCE EFFECT

$\eta_{ISP_{MIX}}$ = MIXING EFFICIENCY

$\eta_{ISP_{VAP}}$ = VAPORIZATION EFFICIENCY

ΔISP_{BL} = BOUNDARY LAYER LOSS FROM TBL

ΔISP_{STRAT} = STRATIFICATION CORRECTION TO BOUNDARY LAYER LOSS

TABLE 15. OME PERFORMANCE SUMMARY

NTO/MMH

REGEN/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s, sec
6000	125	1.65	52.0	80	11	.020	333.6	313.8
		1.45			12	.020	327.6	309.2
		1.85			10	.020	337.0	315.3
	100	1.65	58.5		11	.020	332.9	313.1
		1.45			12	.020	327.1	308.6
		1.85			10	.022	330.0	314.0
	150	1.65	48.0		11	.020	334.1	314.4
		1.45			12	.020	328.0	309.6
		1.85			10	.020	337.8	316.0
	200	1.65	41.5		11	.020	334.8	315.1
		1.45			12	.020	328.4	310.1
		1.85			10	.020	338.7	317.0
	125	1.65	37.0	40	11	.020	326.4	307.7
		1.45			12	.020	321.1	303.7
		1.85			10	.020	329.3	308.7
	100	1.65	42.0		11	.020	325.7	307.0
		1.45			12	.020	320.6	303.2
		1.85			10	.022	328.3	307.4
	150	1.65	34.0		11	.020	326.9	308.2
		1.45			12	.020	321.4	304.0
		1.85			10	.020	329.9	309.2

TABLE 15. OME PERFORMANCE SUMMARY (Concluded)

NTO/MMH

REGEN/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s, sec
6000	200	1.65	30.0	40	11	.020	327.6	308.9
		1.45			12	.020	321.9	304.5
		1.85			10	.020	330.9	310.3
	125	1.65	63.5	120	11	.020	337.1	316.6
		1.45			12	.020	330.7	311.6
		1.85			10	.020	340.8	318.3
	100	1.65	71.0		11	.020	336.3	315.7
		1.45			12	.020	330.2	311.1
		1.85			10	.022	339.7	317.0
	150	1.65	58.0		11	.020	337.5	317.1
		1.45			12	.020	331.0	312.0
		1.85			10	.020	341.5	319.0
	200	1.65	50.0		11	.020	338.2	317.9
		1.45			12	.020	331.5	312.5
		1.85			10	.020	342.5	320.0
3500	125	1.65	40.5	80	9	.036	333.1	310.5
		1.45			10	.033	327.1	307.3
		1.85			8	.037	336.5	310.0
10000		1.65	67.0		12	.020	334.0	314.8
		1.45			13	.020	328.0	310.2
		1.85			11	.020	337.4	316.9
8000		1.65	60.0		12	.020	333.8	314.5
		1.45			13	.020	327.8	309.8
		1.85			11	.020	337.2	316.2

TABLE 16. OME PERFORMANCE SUMMARY

NTO/50-50

REGEN/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s, sec
6000	125	1.6	52.0	80	11	.029	336.2	314.6
		1.4			12	.024	331.2	312.1
		1.8			10	.032	338.3	312.9
	100	1.6	58.5		11	.037	335.4	312.9
		1.4			12	.034	330.6	311.1
		1.8			10	.041	337.2	310.0
	150	1.6	48.0		11	.023	336.8	315.7
		1.4			12	.020	331.6	312.5
		1.8			10	.027	339.2	314.6
	200	1.6	41.5		11	.020	337.6	316.8
		1.4			12	.020	332.1	313.1
		1.8			10	.020	340.4	316.9
	125	1.6	37.0	40	11	.029	328.8	308.3
		1.4			12	.024	324.4	306.4
		1.8			10	.032	330.3	306.1
	100	1.6	42.0		11	.037	328.0	306.6
		1.4			12	.034	323.8	305.3
		1.8			10	.041	329.2	303.1
	150	1.6	34.0		11	.023	329.3	309.2
		1.4			12	.020	324.7	306.7
		1.8			10	.027	331.2	307.8

TABLE 16. OME PERFORMANCE SUMMARY (Continued)

NTO/50-50

REGEN/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s, sec
6000	200	1.6	30.0	40	11	.020	330.1	310.4
		1.4			12	.020	325.3	307.3
		1.8			10	.020	332.3	309.9
	125	1.6	63.5	120	11	.029	339.8	317.5
		1.4			12	.024	334.4	314.6
		1.8			10	.032	342.2	316.0
	100	1.6	71.0		11	.037	339.0	315.9
		1.4			12	.034	333.8	313.6
		1.8			10	.041	341.0	313.1
	150	1.6	58.0		11	.023	340.6	318.6
		1.4			12	.020	334.8	315.0
		1.8			10	.027	343.1	317.8
	200	1.6	50.0		11	.020	341.2	319.7
		1.4			12	.020	335.3	315.5
		1.8			10	.020	344.3	320.0
3500	125	1.6	40.5	80	9	.064	335.7	307.8
		1.4			10	.062	330.7	308.0
		1.8			8	.062	337.8	303.4
10000		1.6	67.0		12	.020	336.7	316.8
		1.4			13	.020	331.7	313.5
		1.8			11	.020	338.8	316.4

TABLE 16. OME PERFORMANCE SUMMARY (Concluded)

NTO/50-50

REGEN/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s, sec
8000	125	1.6	60.0	80	12	.020	336.4	316.4
		1.4			13	.020	331.4	313.4
		1.8			11	.020	338.5	316.1

TABLE 17. OME PERFORMANCE SUMMARY

O_2/MMH

REGEN/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s, sec
6000	100	1.2	58.5	80	12	.026	362.7	334.4
		1.0			14	.027	354.0	330.1
		1.4			10	.022	364.0	331.3
	75	1.2	67.0		12	.044	360.9	330.3
		1.0			14	.044	352.9	328.4
		1.4			10	.036	361.6	325.9
	125	1.2	52.0		12	.020	363.9	336.2
		1.0			14	.020	354.8	331.2
		1.4			10	.020	365.6	333.1
	200	1.2	41.5		13	.020	366.0	339.4
		1.0			15	.020	356.1	333.1
		1.4			11	.020	368.8	337.7
	100	1.2	42.0	40	12	.026	354.0	327.0
		1.0			14	.027	346.6	323.9
		1.4			10	.022	354.4	323.1
	75	1.2	48.5		12	.044	352.3	322.9
		1.0			14	.044	345.5	322.1
		1.4			10	.036	352.2	317.8
	125	1.2	37.0		12	.020	355.2	328.9
		1.0			14	.020	347.3	325.0
		1.4			10	.020	356.0	325.0

TABLE 17. OME PERFORMANCE SUMMARY (Continued)

O_2/MMH

REGEN/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s, sec
6000	200	1.2	30.0	40	13	.020	357.2	331.8
		1.0			15	.020	348.6	326.7
		1.4			11	.020	358.9	329.1
	100	1.2	71.0	120	12	.026	366.8	337.7
		1.0			14	.027	357.6	333.0
		1.4			10	.022	368.6	334.9
	75	1.2	82.0		12	.044	365.0	333.5
		1.0			14	.044	356.4	331.1
		1.4			10	.036	366.1	329.5
	125	1.2	63.5		12	.020	368.0	339.5
		1.0			14	.020	358.3	334.0
		1.4			10	.020	370.2	336.8
	200	1.2	50.0		13	.020	370.2	342.7
		1.0			15	.020	359.7	335.9
		1.4			11	.020	373.6	341.6
3500	100	1.2	45.0	80	10	.050	361.5	327.0
		1.0			12	.067	352.8	324.8
		1.4			8	.033	362.8	323.1
10000		1.2	75.0		14	.020	363.4	338.1
		1.0			16	.020	354.7	332.5
		1.4			12	.020	364.7	335.3

TABLE 17. OME PERFORMANCE SUMMARY (Concluded)

O_2/MMH

REGEN/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s. sec
8000	100	1.2	67.5	80	13	.020	363.2	336.8
		1.0			15	.020	354.5	331.5
		1.4			11	.020	364.5	333.7

TABLE 18. OME PERFORMANCE SUMMARY

O_2/50-50

REGEN/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s. sec
6000	100	1.2	58.5	80	10	.027	364.2	331.7
		1.0			12	.031	358.9	331.6
		1.4			8	.022	361.8	324.8
	75	1.2	67.0		9	.031	362.1	327.4
	125	1.2	52.0		11	.027	365.6	334.9
	200		41.5		12	.025	368.2	338.8

TABLE 19. OME PERFORMANCE SUMMARY

O_2/RP-1

REGENERATIVE COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	ODK I_s, sec	Delivered I_s, sec
6000	100	2.5	58.5	80	24	352.0	327.1
		2.3				350.8	325.9
		2.8				349.4	324.6
	75	2.5	67.0			349.6	324.7
		2.3				348.7	323.8
		2.8				346.8	322.1
	125	2.5	52.0			353.7	328.6
		2.3				352.2	327.4
		2.8				351.3	326.5
	200	2.5	41.5			357.0	332.0
		2.3				354.8	329.9
		2.8				355.2	330.2
	100	2.5	42.0	40		342.4	318.8
		2.3				341.8	318.2
		2.8				339.4	315.9
	75	2.5	48.5			340.1	316.5
		2.3				339.9	316.3
		2.8				337.0	313.7
	125	2.5	37.0			344.0	320.4
		2.3				343.1	319.5
		2.8				341.2	317.8

TABLE 19. OME PERFORMANCE SUMMARY (Continued)

O_2/RP-1

REGENERATIVE COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	ODK I_s, sec	Delivered I_s, sec
6000	200	2.5	30.0	40	24	347.1	323.3
		2.3				345.6	321.9
		2.8				344.7	321.2
	100	2.5	71.0	120		356.7	330.8
		2.3				355.2	329.5
		1.8				354.4	328.7
	75	2.5	82.0			354.2	328.4
		2.3				353.1	327.5
		2.8				351.6	326.0
	125	2.5	63.5			358.5	332.5
		2.3				356.6	330.8
		2.8				356.4	330.6
	200	2.5	50.0			361.9	336.0
		2.3				359.3	333.5
		2.8				360.4	334.6
3500	100	2.5	45.0	80		350.9	325.7
		2.3				349.7	324.6
		2.8				348.3	323.2

TABLE 19. OME PERFORMANCE SUMMARY (Concluded)

O_2/RP-1

REGENERATIVE COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	ODK I_s, sec	Delivered I_s, sec
10000	100	2.5	75.0	80	24	353.0	328.2
		2.3				351.8	327.2
		2.8				350.4	325.8
8000		2.5	67.5			352.5	327.7
		2.3				351.3	326.6
		2.8				349.9	325.2

TABLE 20. OME PERFORMANCE SUMMARY

O_2/N_2H_4, O_2/C_3H_8

F = 6000 Lb

D = 50 In.

Regenerative Cooling

L_{cc} = 15 In.

PROPELLANT	CHAMBER PRESSURE, PSIA	MR	ϵ	ODK I_S, SEC	DELIVERED I_S, SEC
O_2/N_2H_4	75	0.8	43	356.0	326.5
O_2/N_2H_4	100	0.8	58	361.4	331.5
O_2/N_2H_4	125	0.8	73	365.2	334.9
O_2/N_2H_4	100	0.7	58	358.1	328.3
O_2/N_2H_4	100	1.0	58	358.1	328.4
O_2/C_3H_8	75	2.6	43	348.8	333.4
O_2/C_3H_8	100	2.6	58	355.2	339.3
O_2/C_3H_8	125	2.6	73	359.6	343.1
O_2/C_3H_8	100	2.4	58	352.9	337.1
O_2/C_3H_8	100	2.8	58	354.7	338.8

TABLE 21. OME PERFORMANCE SUMMARY

NTO/MMH

DUMP/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s, sec
6000	125	1.65	52.0	80	9	.084	333.6	305.5
		1.45			10	.089	327.6	305.2
		1.85			8	.078	337.0	301.8
	100	1.65	58.5		9	.077	332.9	305.9
		1.45			10	.082	327.1	305.1
		1.85			8	.072	336.0	302.3
	150	1.65	48.0		9	.087	334.1	305.5
		1.45			10	.093	328.0	305.3
		1.85			8	.081	337.8	301.9
	200	1.65	41.5	80	8	.085	334.8	305.4
		1.45			10	.099	328.4	305.4
		1.85			7	.077	338.7	301.9
	125	1.65	37.0	40	9	.084	326.4	299.4
		1.45			10	.089	321.1	299.6
		1.85			8	.078	329.3	295.2
	100	1.65	42.0		9	.077	325.7	299.7
		1.45			10	.082	320.6	299.6
		1.85			8	.072	328.3	295.7
	150	1.65	34.0		9	.087	326.9	299.4
		1.45			10	.093	321.4	299.7
		1.85			8	.081	329.9	295.1

TABLE 21. OME PERFORMANCE SUMMARY (Concluded)

NTO/MMH
DUMP/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s, sec
6000	125	1.65	63.5	120	9	.084	337.1	308.3
		1.45			10	.089	330.7	307.5
		1.85			8	.078	340.8	304.8
	100	1.65	71.0		9	.077	336.3	308.5
		1.45			10	.082	330.2	307.5
		1.85			8	.072	339.7	305.3
	150	1.65	58.0		9	.087	337.5	308.2
		1.45			10	.093	331.0	307.6
		1.85			8	.081	341.5	304.9
3500	125	1.65	40.5	80	8	.090	333.1	302.4
		1.45			9	.100	327.1	302.8
		1.85			6	.071	336.5	298.5
10000		1.65	67.0		8	.083	334.0	305.0
		1.45			10	.099	328.0	305.2
		1.85			8	.083	337.4	301.3
8000		1.65	60.0		8	.077	333.8	305.7
		1.45			10	.093	327.8	305.2
		1.85			8	.077	337.2	302.3

TABLE 22. OME PERFORMANCE SUMMARY

NTO/50-50

Dump/Film Cooling

F = 6000 lb

ϵ = 80

Chamber Pressure, psia	MR	Diameter, in	$\dot{W}_c/W_T$	L_{cc}, in	ODK I_s, sec	Delivered I_s, Sec
125	1.6	52.0	.110	6	336.2	292.5
	1.4		.135	8	331.2	297.5
	1.3		.135	8	327.5	298.6
100	1.6	58.5	.103	6	335.4	293.3
	1.4		.127	8	330.6	298.1
	1.3		.127	8	327.0	299.0
150	1.6	48.0	.115	6	336.8	291.5
	1.4		.129	7	331.6	296.9
	1.3		.141	8	327.8	298.0

TABLE 23. OME PERFORMANCE SUMMARY

O_2/MMH

DUMP/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s, sec
6000	100	1.2	58.5	80	9	.086	362.7	319.7
		1.0			10	.094	354.0	320.9
		1.4			9	.086	364.0	313.5
	75	1.2	67.0		9	.082	360.9	317.9
		1.0			10	.088	352.9	320.3
		1.4			9	.082	361.6	312.2
	125	1.2	52.0		9	.090	363.9	320.2
		1.0			10	.098	354.8	321.3
		1.4			9	.090	365.6	314.0
	200	1.2	41.5		9	.096	366.0	321.2
		1.0			10	.106	356.1	322.0
		1.4			9	.096	368.8	315.4
	100	1.2	42.0	40	9	.086	354.0	312.4
		1.0			10	.094	346.6	314.7
		1.4			9	.088	354.4	305.4
	75	1.2	48.5		9	.082	352.3	310.7
		1.0			10	.088	345.5	314.1
		1.4			9	.082	352.2	304.2
	125	1.2	37.0		9	.090	355.2	313.0
		1.0			10	.098	347.3	315.2
		1.4			9	.090	356.0	306.0

TABLE 23. OME PERFORMANCE SUMMARY (Concluded)

O_2/MMH

DUMP/FILM COOLING

Thrust, lb	Chamber Pressure, psia	MR	Diameter, in	ϵ	L_{cc}, in	$\dot{W}_c/\dot{W}_T$	ODK I_s, sec	Delivered I_s, sec
6000	100	1.2	71.0	120	9	.086	366.8	322.9
		1.0			10	.094	357.6	323.7
		1.4			9	.086	368.6	317.2
	75	1.2	82.0		9	.082	365.0	321.1
		1.0			10	.088	356.4	323.0
		1.4			9	.082	366.1	315.8
	125	1.2	63.5		9	.090	368.0	323.4
		1.0			10	.098	358.3	324.1
		1.4			9	.090	370.2	317.6
3500	100	1.2	45.0	80	9	.106	361.5	314.1
		1.0			10	.115	352.8	317.2
		1.4			8	.096	362.8	307.1
8000		1.2	67.5		10	.085	363.2	322.7
		1.0			10	.085	354.5	322.3
		1.4			9	.080	364.5	315.7
10000		1.2	75.0		10	.080	363.4	323.8
		1.0			10	.080	354.7	323.1
		1.4			9	.075	364.7	317.3

TABLE 24. Typical Mixture Ratio and Chamber Pressure Tolerances

NTO/MMH

Regenerative/Film Cooling

Case	Worst Case MR	Worst Case MR, psia
Calibrated Engine	1.65 ± .02	125 ± 2.5
Inlet Pressure Excursions of ± 4 psi oxid, ± 4 psi fuel	1.65 ± .15	125 ± 4.8
Inlet Pressure and Inlet Temperature Exclusions (40-90F, Max ΔT=10F)	1.65 ± .16	125 ± 5.4
Oxid or Fuel Regulator Malfunction and Nominal Inlet Pressure and Temperature exclusions (U/S Regulator + 10 psia	1.65 ± .34	133 (max)
Ball Valve Malfunction (Single Oxid and Fuel Flow Path Closed) Minimum Inlet Pressures and Maximum Inlet Temperature	1.65 ± .05	116.6 (min)

TABLE 25. OPERATING SENSITIVITY COMPARISON

PROPELLANTS COOLING	NTO/MMH Regen	NTO/MMH Film	NTO/50-50 Regen	NTO/50-50 Film	O_2/MMH Regen	O_2/MMH Film	O_2/RP-1 Regen
Specific Impulse							
P_{fuel} inlet, sec/psi	-0.24	0	-0.15	+0.30	-0.4	-0.4	0
$P_{oxidizer}$ inlet, sec/psi	+0.22	-.04	+0.13	-0.35	+0.2	+0.4	-0.15
P_{inlet} (both), sec/psi	-0.02	0	-0.02	0	+0.1	0	0
W_c, sec/percent	-0.5	-1.9	-1.0	-2.7	-1.3	-1.0	-
Thrust							
P_{fuel} inlet, lb/psi	+7.9	+9.1	+7.2	+9.1	+11.4	+15.0	+8.3
$P_{oxidizer}$ inlet, lb/psi	+18.2	+17.8	+18.7	+17.3	+19.5	+18.0	+24.6
P inlet (both), lb/psi	+26	+27	+26	+26	+31	+33	+33

TABLE 26

OME COMPLEXITY COMPARISON

CHAMBER COOLING CONCEPT

FILM COOLED	REGENERATIVE COOLED
CHANNELED SLEEVE IN CHAMBER (S)	CHANNELED DOUBLE WALL CHAMBER (C)
HOT WALL INSULATION (C)	COOL THROAT AND WALL (S)
PROPELLANT MANIFOLD CONFIGURATION (S)	PROPELLANT MANIFOLD CONFIGURATION (C)
ALL METAL CONSTRUCTION (S)	ALL METAL CONSTRUCTION (S)
FABRICATION -- MACHINING, WELDING, COATING, ASSEMBLY (S)	FABRICATION -- MACHINING, ELECTROFORM, WELDING, ASSEMBLY (C)
HANDLING LARGE SINGLE PIECE (C)	HANDLING TWO PIECE CHAMBER (S)
SEVEN JOINTS (S)	THIRTEEN JOINTS (C)
SMALL COOLANT CHANNEL SIZES -- CRITICAL (C)	LARGE CHANNELS AND SMALL BLC ORIFICES -- NOT CRITICAL (S)
DIFFICULT TO CLEAN SMALL ORIFICES (C)	EASY TO CLEAN CHANNELS (S)

SIMPLICITY (S)

COMPLEXITY (C)

TABLE 26 (CONTINUED)
OME RELIABILITY COMPARISON

CHAMBER COOLING CONCEPT

FILM COOLED	REGENERATIVE COOLED
FEW JOINTS, FEW PARTS	FEW PARTS, MORE JOINTS, MORE MANIFOLDS
SINGLE HOT WALL CHAMBER	DOUBLE WALL CHAMBER, COOL THROAT AND WALL
CRITICAL SMALL FILM ORIFICES	LARGE CHANNELS, BLC ORIFICES NOT CRITICAL
COATING ON CHAMBER WALL	NO THROAT COATING, REPLACEABLE NOZZLE COATED
HOT WALL INSULATION	NO INSULATION
HIGH TEMPERATURE COLUMBIUM CHAMBER	STANDARD 321 STAINLESS STEEL CHAMBER
HIGH THERMAL CYCLES	MODEST THERMAL CYCLES
SENSITIVE TO MECHANICAL DAMAGE	COMBUSTOR INSENSITIVE TO LIGHT MECHANICAL DAMAGE
	WIDER MIXTURE RATIO OPERATING RANGE

TABLE 26 (CONCLUDED)

OME SAFETY COMPARISON

CHAMBER COOLING CONCEPT

FILM COOLED	REGENERATIVE COOLED
SINGLE HOT WALL	DOUBLE WALL REDUNDANCY
HOT WALL FAILURE CATASTROPHIC	HOT WALL FAILURE SELF COOLING
COATING SENSITIVE TO DAMAGE	COATING IN NOZZLE ONLY
MINIMUM NUMBER OF LEAK PATHS	MORE JOINTS
CONTINUOUS SINGLE WALL	TWO PIECE CHAMBER-NOZZLE JOINT
SMALL CRITICAL COOLANT CHANNELS	LARGE CHANNEL PASSAGES

TABLE 27

OME MAINTAINABILITY COMPARISON

CHAMBER COOLING CONCEPT

FILM COOLED	REGENERATIVE COOLED
ALL METAL CONSTRUCTION	ALL METAL CONSTRUCTION
LARGE SINGLE UNIT	TWO PIECE, EASILY HANDLED CHAMBER-NOZZLE
NOZZLE DAMAGE REQUIRES CHAMBER REPLACEMENT	NOZZLE REPLACEABLE IN VEHICLE
COATED COMBUSTOR AND NOZZLE	UNCOATED COMBUSTOR AND COATED NOZZLE
COMBUSTOR AND COMPONENT INSULATION	NO INSULATION
MANIFOLD EASILY PURGED	MORE MANIFOLDS TO PURGE

TABLE 28

OME COMPLEXITY COMPARISON

PROPELLANTS

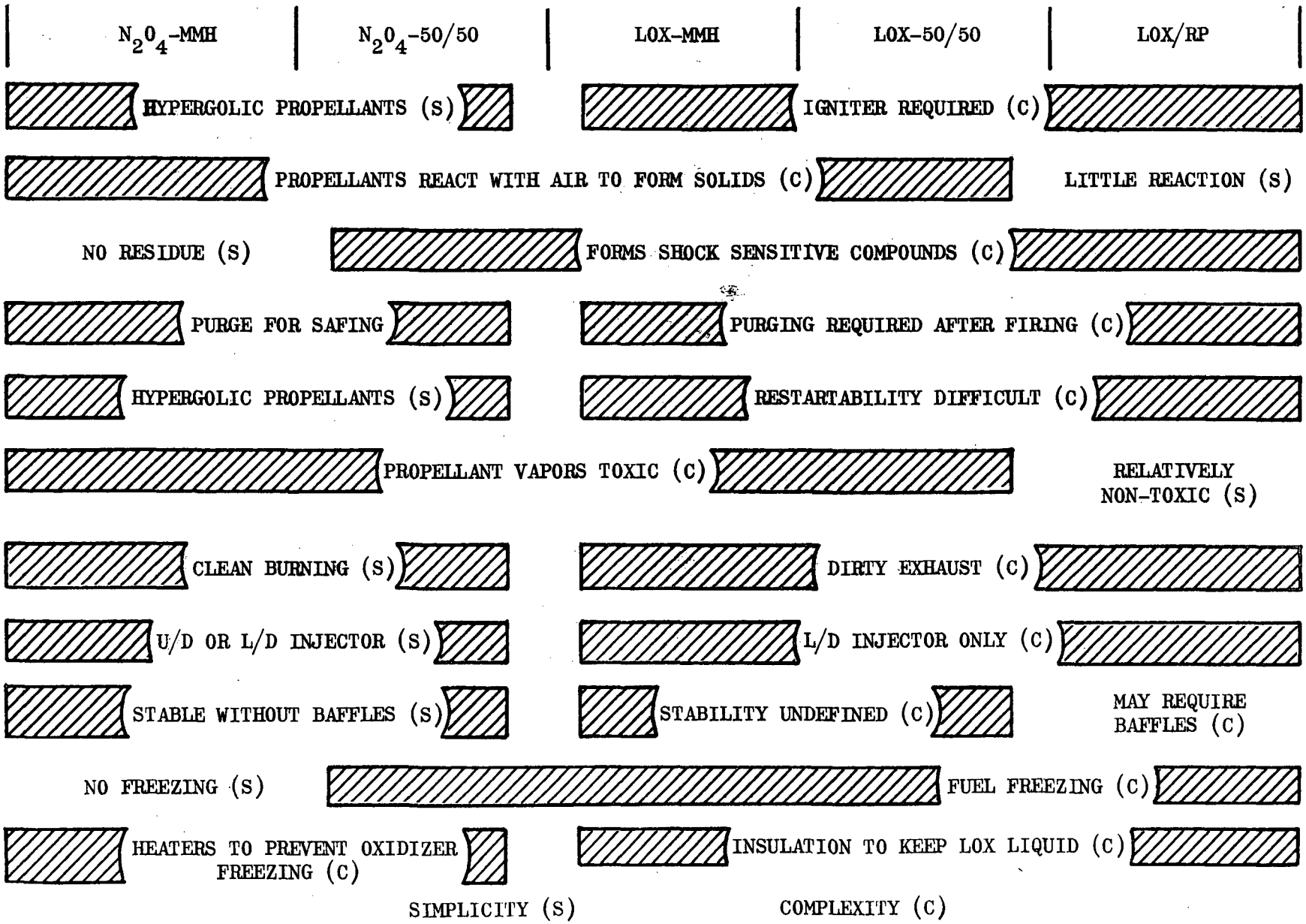

TABLE 29

OME RELIABILITY COMPARISON
PROPELLANTS

N_2O_4-MMH	N_2O_4-50/50	LOX-MMH	LOX-50/50	LOX-RP
HYPERGOLIC PROPELLANTS		IGNITER REQUIRED		
REQUIRES PRESSURE CONTROLS		REQUIRES PRESSURE AND TEMPERATURE CONTROLS		
PROPELLANTS DEGRADE ON AIR CONTACT-PURGING REQUIRED		FUEL DEGRADES ON AIR CONTACT -PURGING REQUIRED		DEGRADATION VERY SLOW
PURGE FOR SAFING		PURGING REQUIRED AFTER EACH FIRING		
NO RESIDUE	FORMS SHOCK SENSITIVE COMPOUNDS			
HEATERS FOR OXIDIZER	HEATERS FOR OXIDIZER AND FUEL	INSULATION TO KEEP LOX LIQUID	HEATERS FOR FUEL	
TEMPERATURE VARIATIONS SMALL		TEMPERATURE VARIATIONS LARGE-MAY CAUSE LEAKAGE		
TECHNOLOGY WELL DEVELOPED		LITTLE TECHNOLOGY		LIMITED TECHNOLOGY

TABLE 30

OME SAFETY COMPARISON
PROPELLANTS

N_2O_4-MMH	N_2O_4-50/50	LOX-MMH	LOX-50/50	LOX-RP
FILTRATION SATISFACTORY	PROPELLANT SYSTEM CLEANLINESS CRITICAL			
STABLE PROPELLANT	FUEL CATALYTICALLY DECOMPOSES	STABLE PROPELLANT	FUEL CATALYTICALLY DECOMPOSES	STABLE PROPELLANT
NO RESIDUE	NON-HYPERGOLIC REACTION PRODUCTS SHOCK SENSITIVE			
FUELS IGNITEABLE BY ELECTRICAL SPARK, FLAME, MECHANICAL SHOCK				
FUEL READILY FLAMMABLE				LEAST FLAMMABLE
TOXIC OXIDIZER		FROSTBITE, SPONTANEOUS IGNITION WITH FUELS		
FUELS VERY TOXIC				IRRITABLE
SPILLS FLAMMABLE		SPILLS-FLAMMABLE AND EXPLOSIVE		
DILUTE WITH WATER		DILUTE WITH CO_2, FOAM, FOG		

TABLE 31

OME MAINTAINABILITY COMPARISON

PROPELLANTS

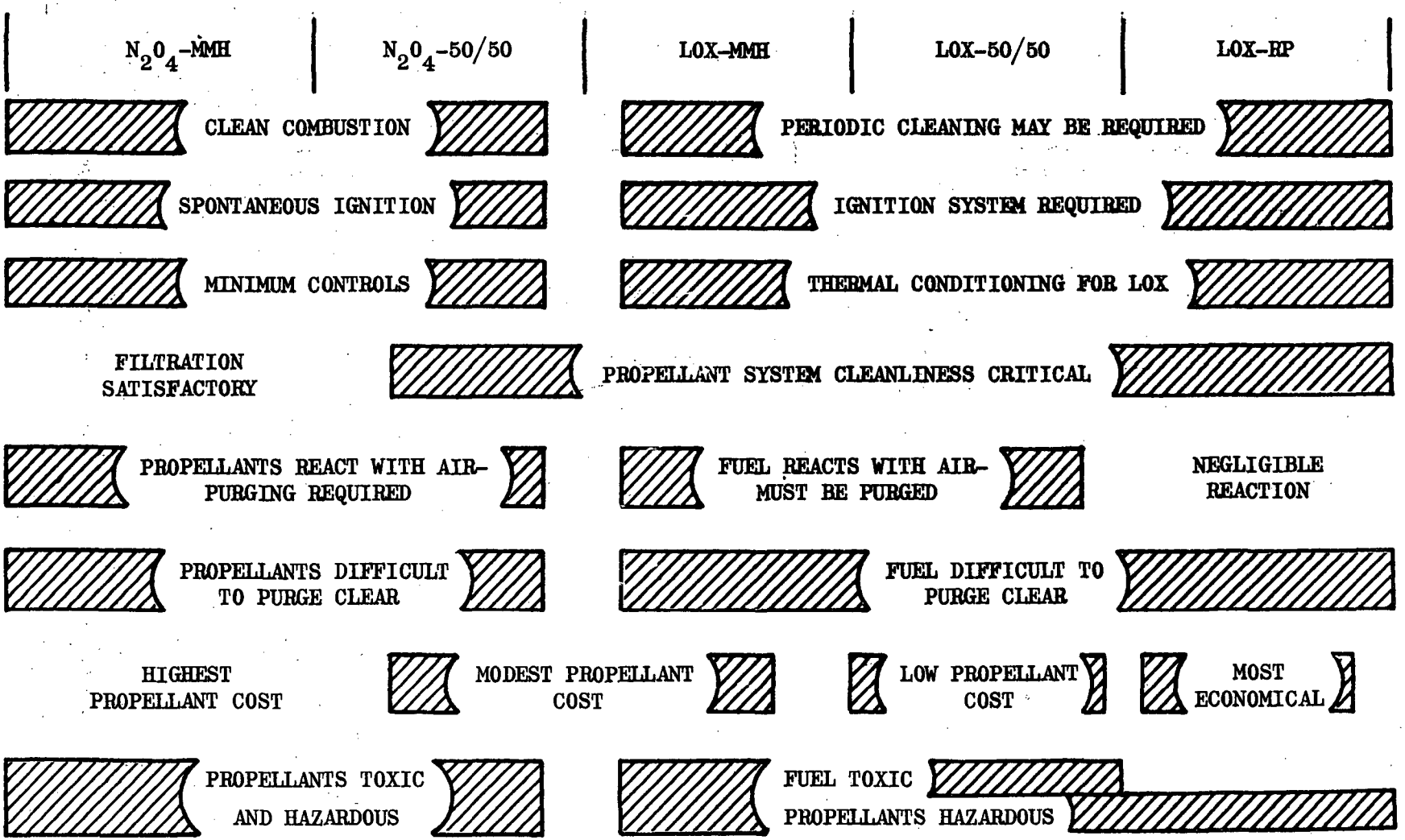

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TABLE 32. OME COMBUSTION PRODUCTS

PROPELLANT	NTO/MMH	NTO/50-50	LOX/MMH	LOX/50-50	LOX/RP-1
O/F	1.65	1.6	1.2	1.2	2.5
H_2O	0.293	0.321	0.369	0.406	0.259
CO_2	0.109	0.102	0.151	0.150	0.399
CO	0.160	0.114	0.180	0.117	0.318
N_2	0.417	0.443	0.273	0.300	
$H+H_2$	0.017	0.014	0.019	0.014	0.011
NO	0.004	0.006	0.007	0.011	
$O+O_2$			0.001	0.002	0.011
OH					0.002
COMPARATIVE RATING	NOMINAL	WORSE	WORSE	WORSE	BETTER

TABLE 33. TECHNOLOGICAL AND OPERATIONAL COMPARISON OF COOLING METHODS

Cooling Method	Complexity	Reliability	Safety	Maintainability	Development Risk	Ecology	Overall
Regenerative Film	Simpler	More Reliable	Safer	Same	Lower Risk	Same	Better

TECHNOLOGICAL AND OPERATIONAL COMPARISON OF PROPELLANT COMBINATIONS

Propellants	Complexity	Reliability	Safety	Maintainability	Development Confidence	Ecology	Overall
NTO/MMH	Simplest	Most Reliable	Safest	Most Maintainable	Lowest Risk	2	Best
NTO/50-50	2	2	2	Most Maintainable	Lowest Risk	3	2
O_2/MMH	4	4	3	3	3	3	4
O_2/50-50	5	5	4	3	3	4	5
O_2/RP-1	3	3	2	2	2	Least Contaminants	3

TABLE 34. SUMMARY OF DEVELOPMENT PROGRAM PROPELLANT COST

PROPELLANT	COST	N_2O_4/MMH RE-GENERATIVE & FILM T/C	N_2O_4/50-50 REGENERATIVE& FILM T/C	LOX/RP-1 REGENERATIVE T/C	LOX/MMH REGENERATIVE T/C	LOX/50-50 REGENERATIVE T/C	LOX/MMH FILM T/C
NTO	$0.14/Lb	75,600	74,760				
MMH	$2.50/Lb *	817,500			973,200		999,700
50-50	$0.75/Lb		250,500			286,110	
LOX	$26/Ton			9,475	8,071	8,183	8,291
RP-1	$.076/Lb			20,011			
GN_2	$1.60/KSCF	296,864	296,864	317,644	326,550	326,550	335,456
Helium	$45/KSCF	455,625	455,625	540,000	546,675	546,675	560,420
LN_2	$25/Ton	7,000	7,000	8,500	8,750	8,750	8,750
LOX	$26/Ton	479,362	479,362	503,330	508,125	508,125	527,300
Isopropyl Alc	$0.05/Lb	553,550	553,550	581,228	586,760	586,760	608,905
H_2O_2	$0.30/Lb	981	1,002	--	1,168	1,145	1,200
TOTAL		$2,686,482	$2,118,663	$1,980,188	$2,959,300	$2,272,300	$3,050,000

* $1.25/lb for Flight Program

TABLE 35. ROUTINE OME MAINTENANCE COSTS

	NTO/MMH		LOX/MMH		LOX/RP-1	
	LAB $	MAT $	LAB $	MAT $	LAB $	MAT $
Prelim. Inspection	17		21		21	
Data Analysis	144		162		162	
GN_2 Purge	76	20	102	27	102	31
Visual Inspection	68		77		77	
Cov./Clos. Installation	17		17		17	
Flush					136	35
TOTAL	322	20	379	27	515	66

TABLE 36. IN VEHICLE CORRECTIVE MAINTENANCE DELTA FOR NON HYPERGOLIC PROPELLANT OME

Problem	Freq/ 1000 MDC	Action	Man Hours	Man Hrs & Cost/ 1000 MDC	Material Consumed	Mat'l Cost	Mat'l Cost/ 1000 MDC
Purge Valve	3.4	Replace	15	51.0/$867	1 $ft^3 GN_2$	1	3
					Interf Seals	6	20
Position Indicators	2.0	Replace	23.0	46.0/$782	1 $ft^3 GN_2$	1	2
					Interf Seals	4	8
Injector & Nozzle	1.3	Drains Purge System	12	15.6/$265	40 CuFt	30	78
					Interf Seals	7	18
		Replace	26	33.7/$573			
Bi Prop Valve	1.3	Replace	12	15.6/$265	40 FT^3	30	39
			30	39/$510	Interf	7	9
Spark Plug & Electrical	4.0	Replace	8	32/$544	10 CuFt	7	28
					Interf Seal	3	12
TOTAL				$3806			$217

TABLE 37. SITE-SHOP CORRECTIVE MAINTENANCE FOR OME

Item	Freq/ 1000 MDC	Action	Man Hours	Man Hrs & Cost 1000 MDC	Mat'l Consumed	Mat'l Cost, $	Mat'l Cost/ 1000 MDC, $
Repair Purge Valve	3.4	Pack & Return to Factory	2.0	6.8/$116	Pack	3	11
Position Indicators	2.0	Pack & Return to Factory	2.0	4.0/$68	Pack	3	6
Injector and Nozzle	1.3	Decontam.	1.0	13/$210	7 Gal Alc 7 Gal Fr 12 ft^3 GH_2	11 50 10	14 65 13
		Pack & Return	3.0	3.9/$66	Pack	3	4
Bi Prop Valve	1.3	Decontam.	1.3	16.9/$287	10 Gal Alc 10 Gal Fr 15cu ft GN_2	15 65 12	20 85 15
		Pack & Return	4.0	5.2/$89	Pack	3	4
Spark Plug & Electrical	4.0	Pack & Return	3.0	14/$204	Pack	3	12
				$1040			$249

TABLE 38. FACTORY CORRECTIVE MAINTENANCE FOR NON-HYPERGOLIC OME

Item	Freq/ 1000	Action	Man Hours	Man Hrs & Cost	Mat'l Consumed	Mat'l Cost, $	Mat Cost/ 1000 MDC, $
Repair Purge Valve	3.4	Inspect, Replace Package	$1000+ 4 Hrs	1000 MDC /$3,400 13.6/$231	Pack.	4	13
Position Indicator	2.0	Inspect, Replace Package	$700+ 4 Hrs	/$1,400 8/$136	Pack.	4	8
Inject & Nozzle	1.3	Inspect, Replace Package	$10,000 +4 Hrs	/$13000 5.2/$89	Pack.	200 4	260 5
Bi Prop Valve	1.3	Inspect, Replace Package	$ 3,000 +4 Hrs	/$3900 5.2/$89	Pack.	4	5
Spark Plug & Electrical	4.0	Inspect, Replace Package	$ 3,000	/$12000 16/$272	Pack.	100 4	400 16
				$ 34,517			$707

TABLE 39. DELTA HARDWARE TRANSPORTATION COSTS FOR CORRECTIVE MAINTENANCE OF NONHYPERGOLIC PROPELLANT OME

Item	Freq/ 1000 MDC	Transportation Requirement	Method	Cost	Cost/1000 MDC
Repair Purge Valve	3.4	Ship to Factory & Return to Site	Air Freight	$45	$153
Position Indicator	2.0	"	"	$45	90
Injector & Nozzle	1.3	"	"	$45	59
Bipropellant Valve	1.3	"	"	$45	59
Spark Plug & Electrical	4.0	"	"	$45	180
TOTAL					$541

TABLE 40. MAINTENANCE COST SUMMARY FOR OME ORBITER MISSION CYCLE

	N_2O_4/MMH N_2O_4/50-50			LOX/MMH LOX/50-50			LOX/RP-1		
	$	Labor $	Mat'l$		Labor $	Mat'l$		Labor $	Mat'l $
ROUTINE MAINTENANCE		483	20		569	27		773	66
CORRECTIVE MAINTENANCE									
IN-VEHICLE		39	1		50	2		50	2
SITE ENGINE SHOP		14	3		16	4		16	4
FACTORY		158	174		193	175		193	175
TRANSPORTATION	4			5			5		
SUB TOTAL	4	694	198	5	828	208	5	1032	247
TOTAL		$ 896			$1041			$1284	
DELTA TO N_2O_4/MMH		--			$145			$388	
PERCENTAGE DELTA		--			16%			43%	

TABLE 41. OME PROGRAM COST & SCHEDULE COMPARISON SUMMARY
N_2O_4 REGENERATIVE COOLED THRUST CHAMBER AS BASELINE

PROP. COMB.	Δ SCHEDULE		Δ MANPOWER AND HARDWARE COST		Δ PROPELLANT COST DEVELOPMENT PROGRAM		Δ MAINTENANCE COST PER FLIGHT		Δ COST, SPARES		Δ COST, LOGISTICS	
	REGEN	FILM	REGEN	FILM	REGEN	FILM	REGEN	FILM	REGEN	FILM	REGEN	FILM
N_2O_4/MMH	B.L.	0	B.L.	-$85,000	B.L.	0	B.L.	0	B.L.	0	B.L.	0
N_2O_4/50-50	0	0	0	-$85,000	-$567,780	-$567,780	0	0	0	0	0	0
LOX/MMH	+10 Wks	+12 Wks	$1,346,000	+$1,401,000	+$282,820	+$363,520	$145(16%)	$145(16%)	$58,000	$58,000	$19,000	$19,000
LOX/50-50	+10 Wks		+$1,346,000		-$414,180		$145(16%)		$58,000		$19,000	
LOX/RP-1	+13 Wks		+$1,486,000		-$706,280		$388(43%)		$58,000		$19,000	

TABLE 42. OME PROGRAM COSTS THROUGH 100 FLIGHTS
N_2O_4 REGENERATIVELY COOLED THRUST CHAMBER AS BASELINE

	Δ PROPELLANT COST PER FLIGHT		Δ PROGRAM + SUPPORT COSTS		Δ PROP. & MAINTENANCE COST FOR 100 FLIGHTS		TOTAL Δ COST THRU 100 FLIGHTS	
	REGEN	FILM	REGEN	FILM	REGEN	FILM	REGEN	FILM
N_2O_4/MMH	B.L.	0	B.L.	-$ 85,000	B.L.	0	B.L.	-$ 85,000
N_2O_4/50-50	- 5,105	- 5,105	-$ 567,780	-$ 652,780	-$ 510,500	-$ 510,500	-$1,078,280	-$1,163,280
LOX/MMH	- 1,658	- 1,658	+$1,705,820	+$1,841,520	-$ 151,300	-$ 151,300	+$1,554,520	+$1,690,220
LOX/50-50	- 7,315		+ 1,008,820		-$ 717,000		+$ 291,820	
LOX/RP-1	-14,688		+$ 856,720		-$1,430,000		-$ 573,280	

Figure 1. **REGENERATIVELY COOLED OME**

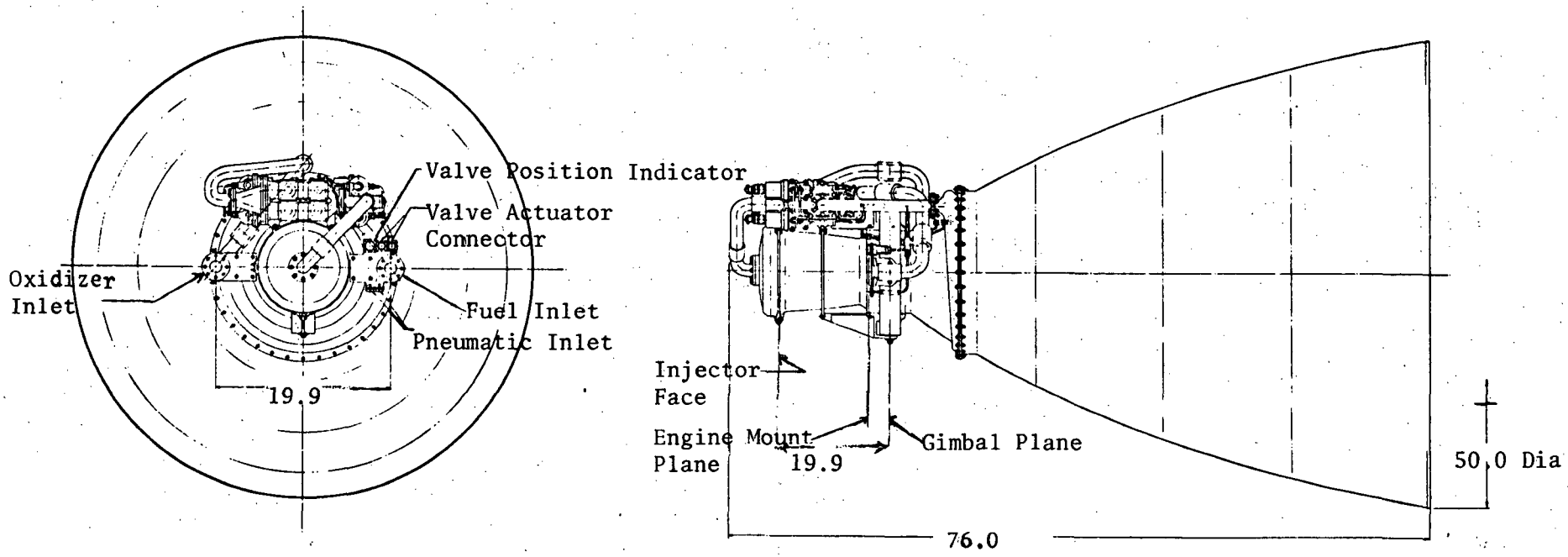
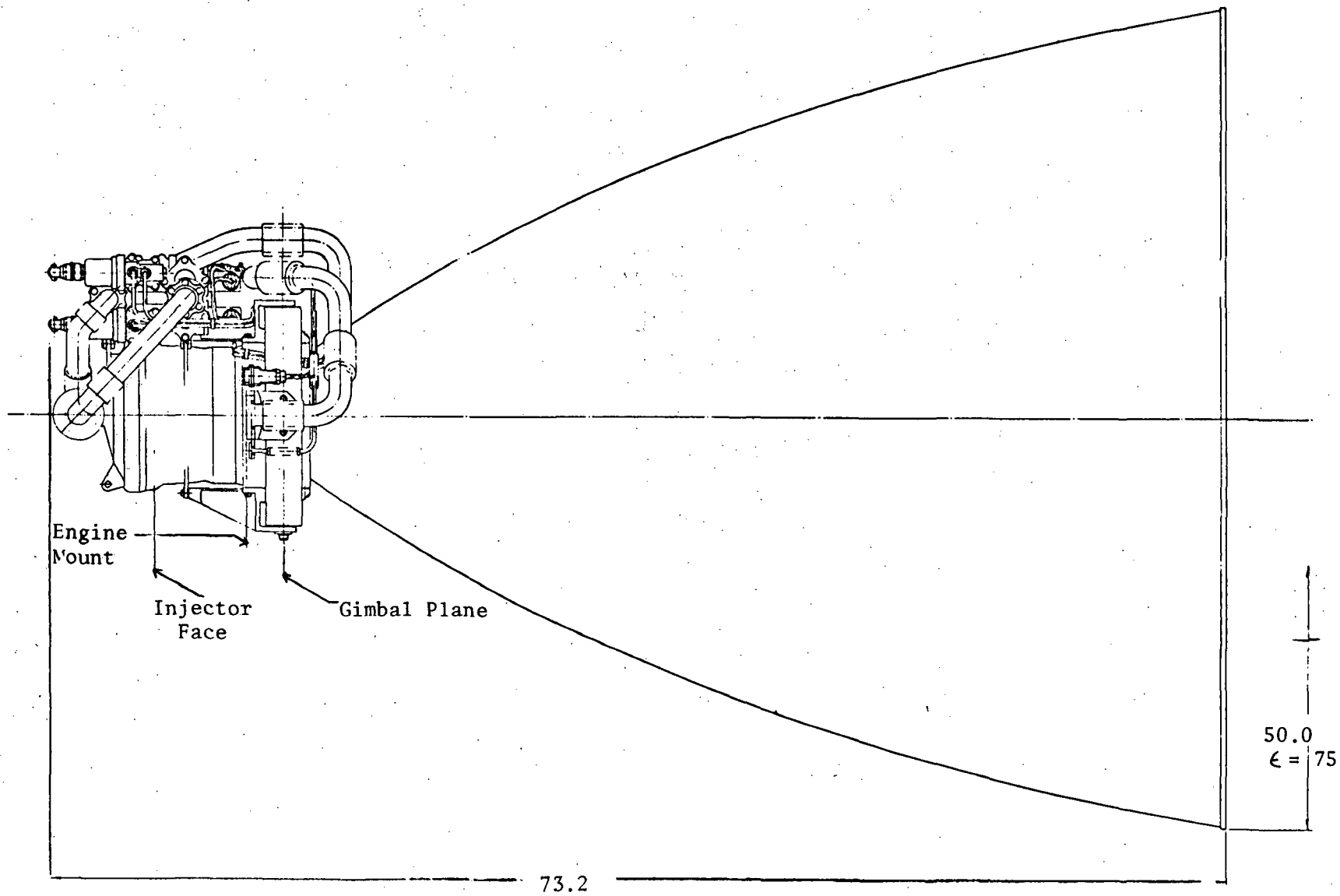


Figure 2. **DUMP/FILM COOLED OME**



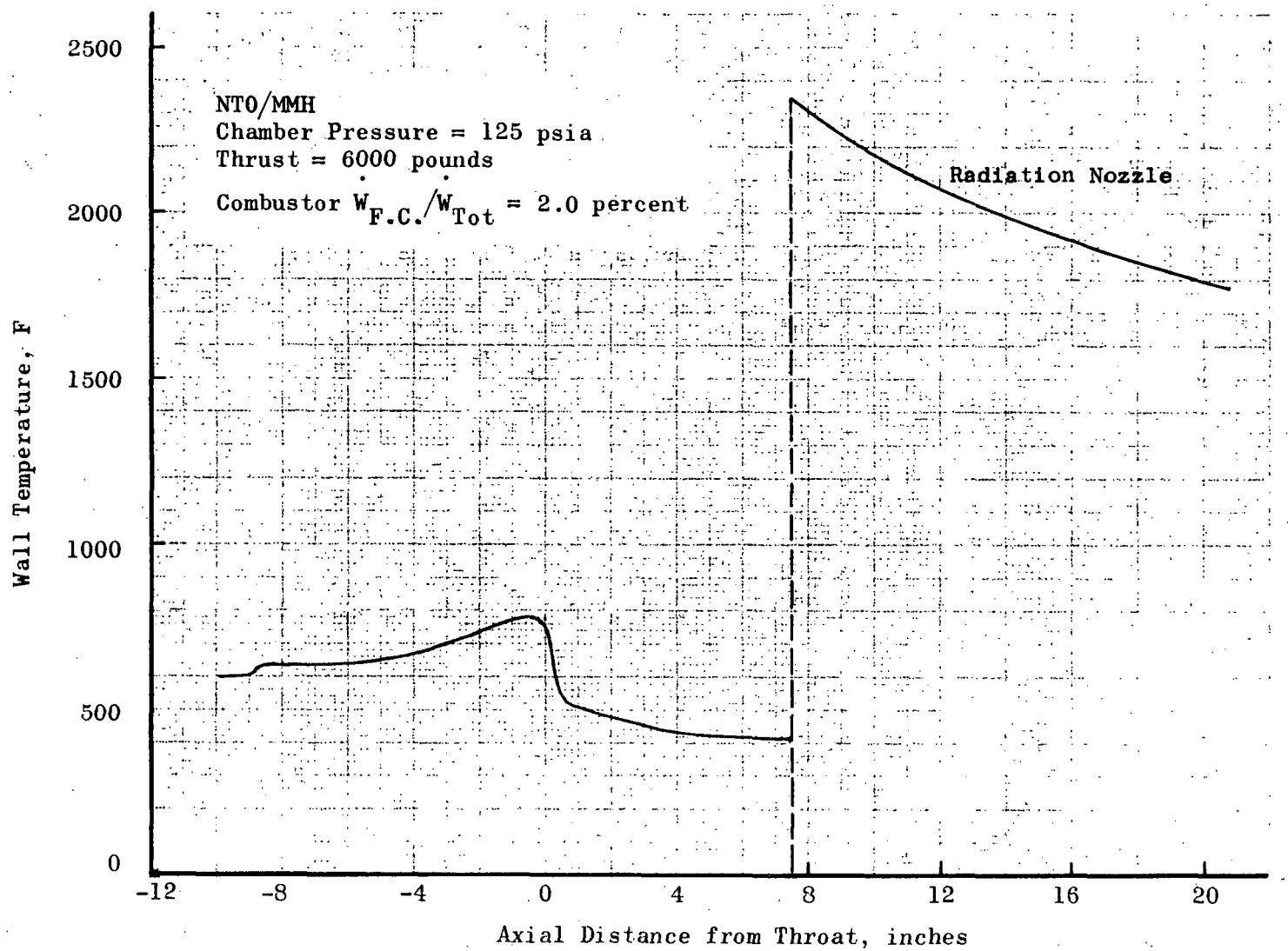


Figure 3. Typical Regeneratively Cooled Hot Wall Temperature Profile

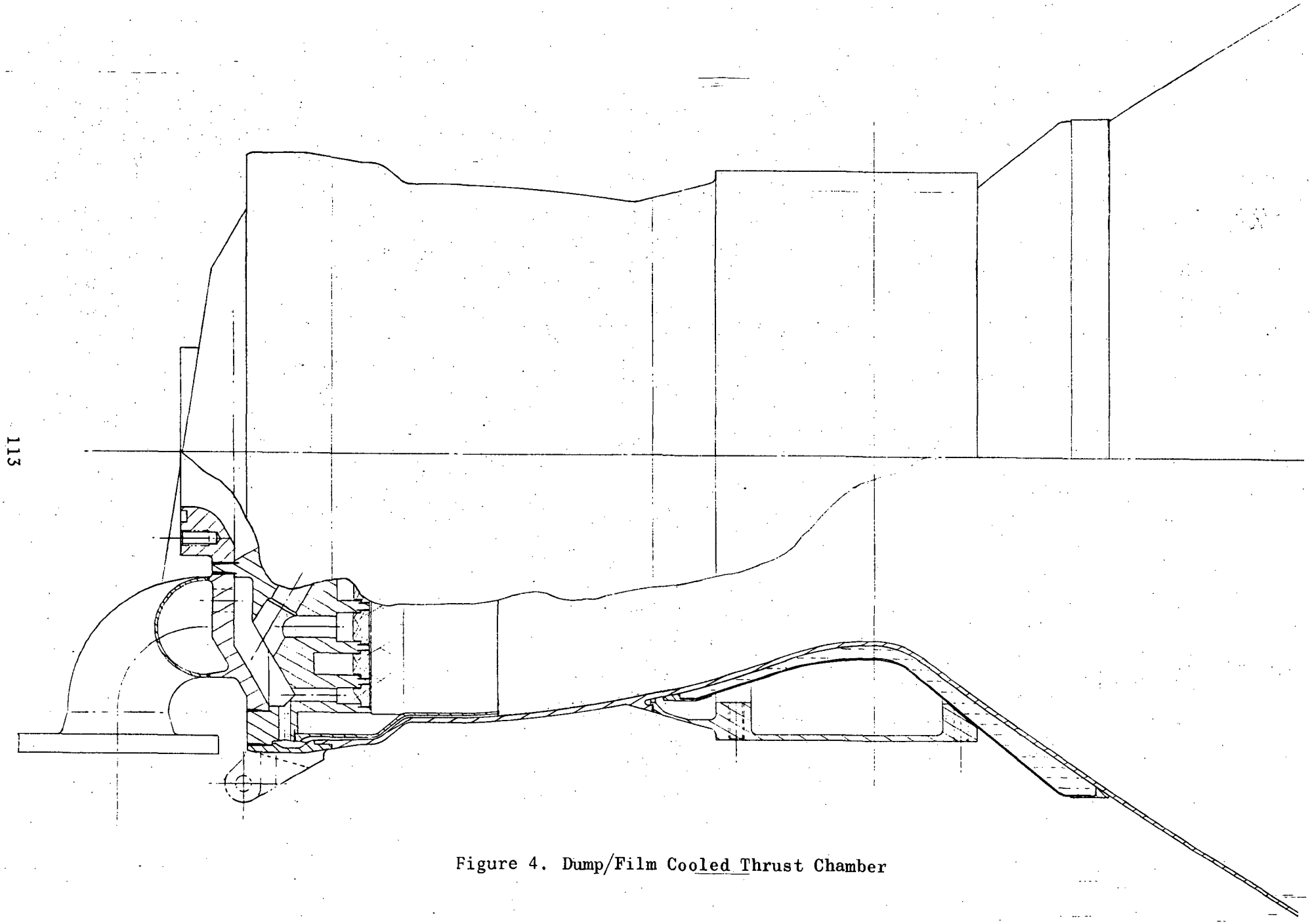


Figure 4. Dump/Film Cooled Thrust Chamber

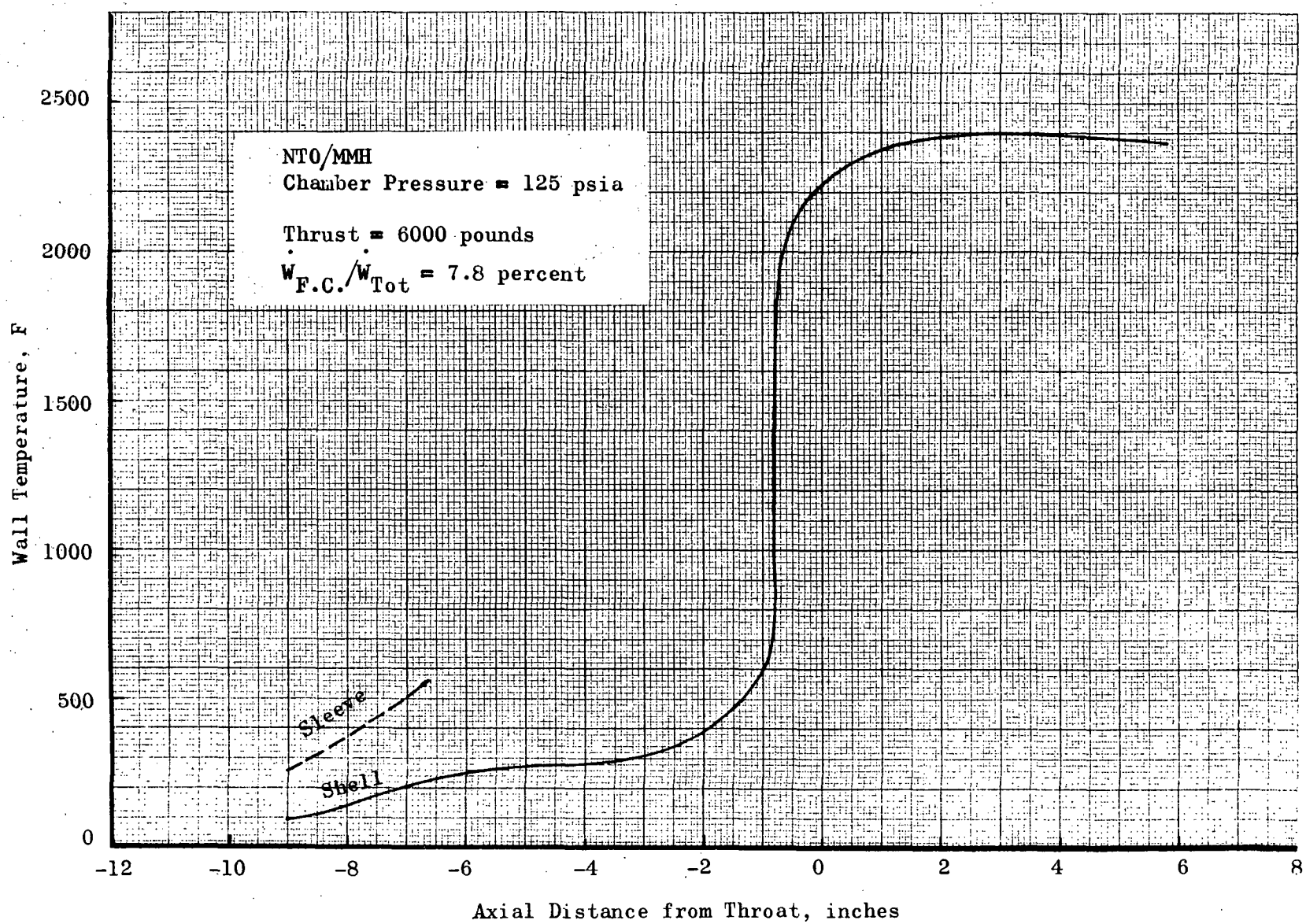
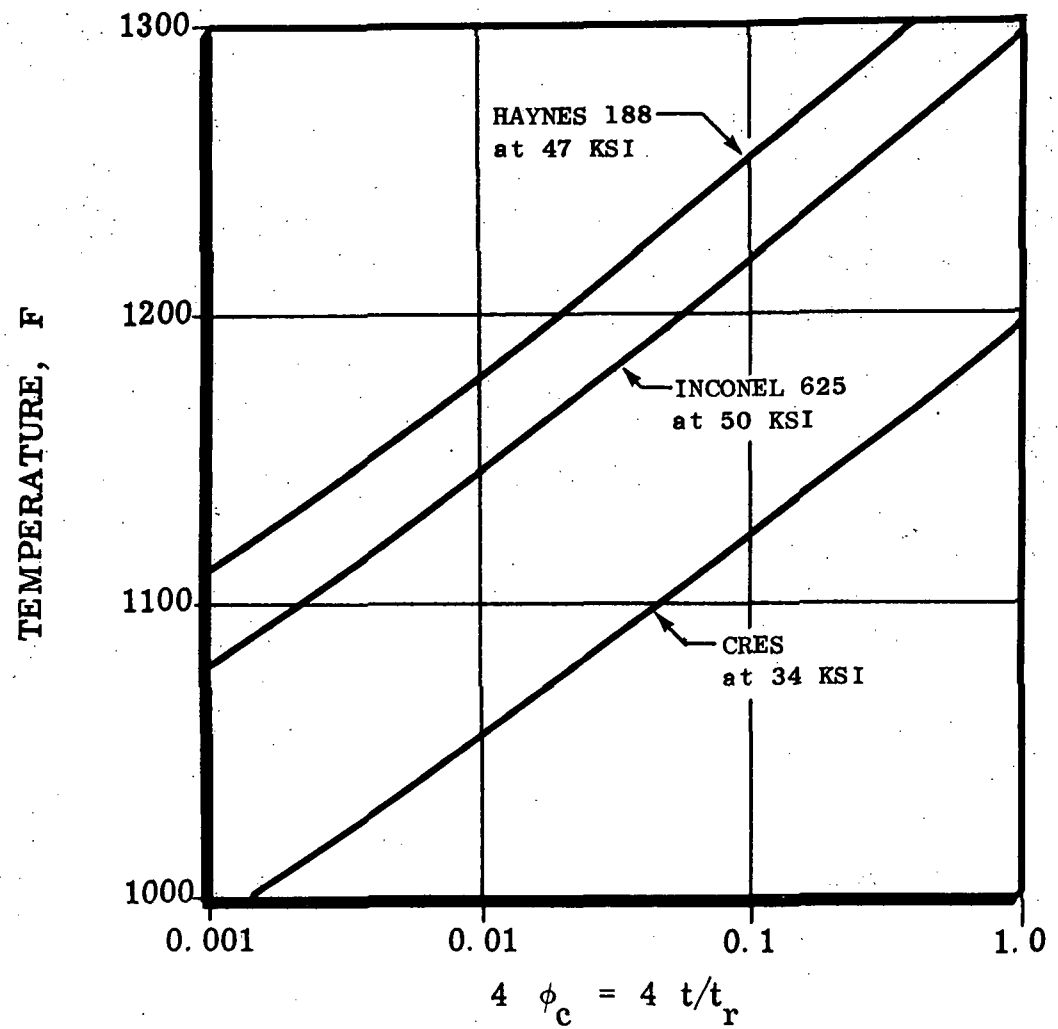


Figure 5. Typical Dump/Film Cooled Chamber Hot Wall Temperature Profile



t = 15 HOURS

t_r = TIME TO RUPTURE

NOTE: THE STRESS LEVELS INDICATED FOR EACH PLOT TYPIFY THE MAXIMUM HOT GAS WALL STRESS. STRESS RELAXATION HAS NOT BEEN INCLUDED.

CONCLUSION: NO SIGNIFICANT CREEP DAMAGE ON REGEN. CHAMBER.

Figure 6. SS/OME Creep Damage Regenerative Cooled Chamber

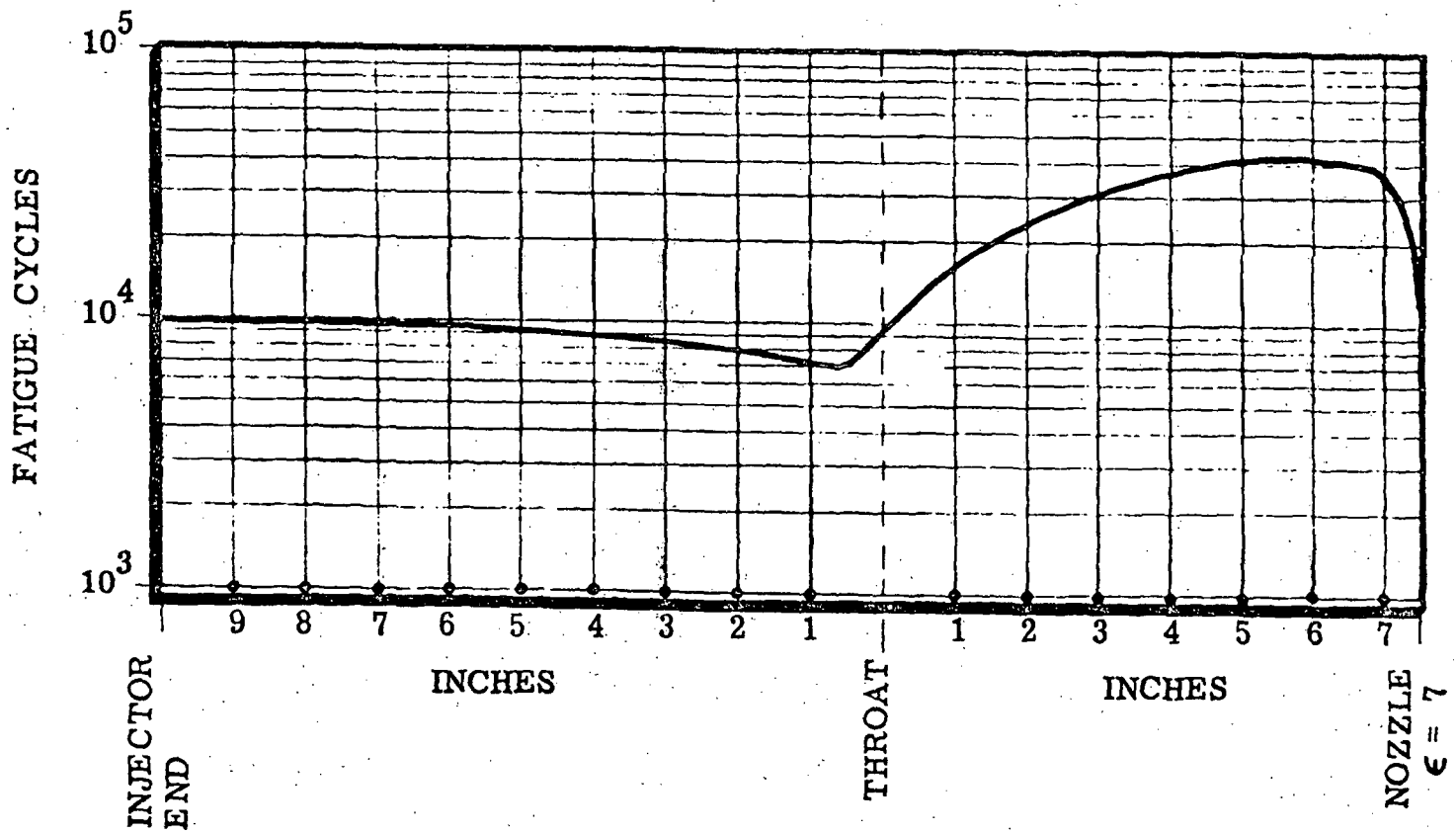


Figure 7. REGENERATIVELY COOLED THRUST CHAMBER CYCLE LIFE CAPABILITY (INCLUDES 60 HOURS OF OPERATION)

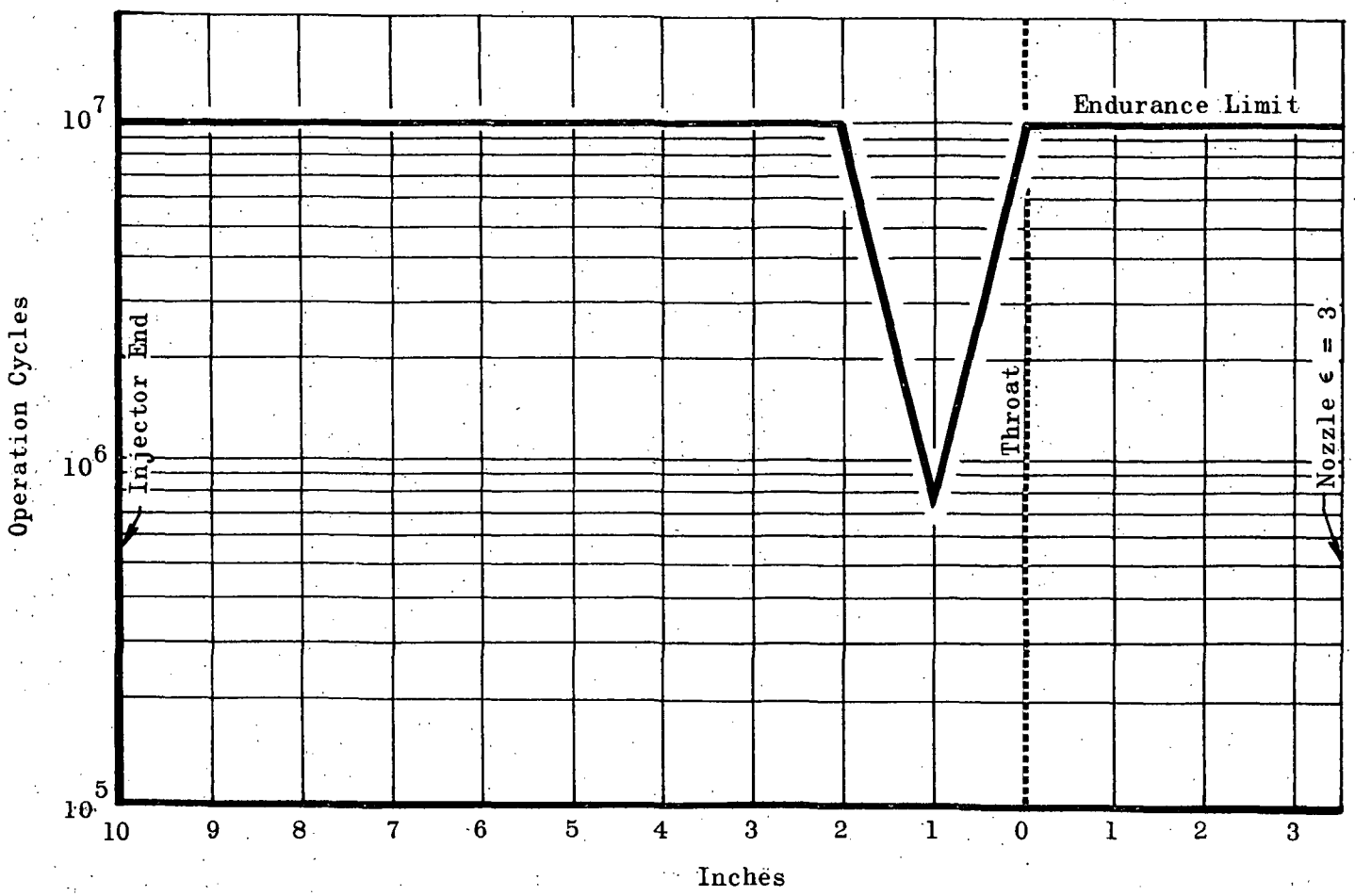


Figure 8. Dump/Film Cooled Thrust Chamber Cyclic Capability
(Includes 60 Hours of Total Operating Time)

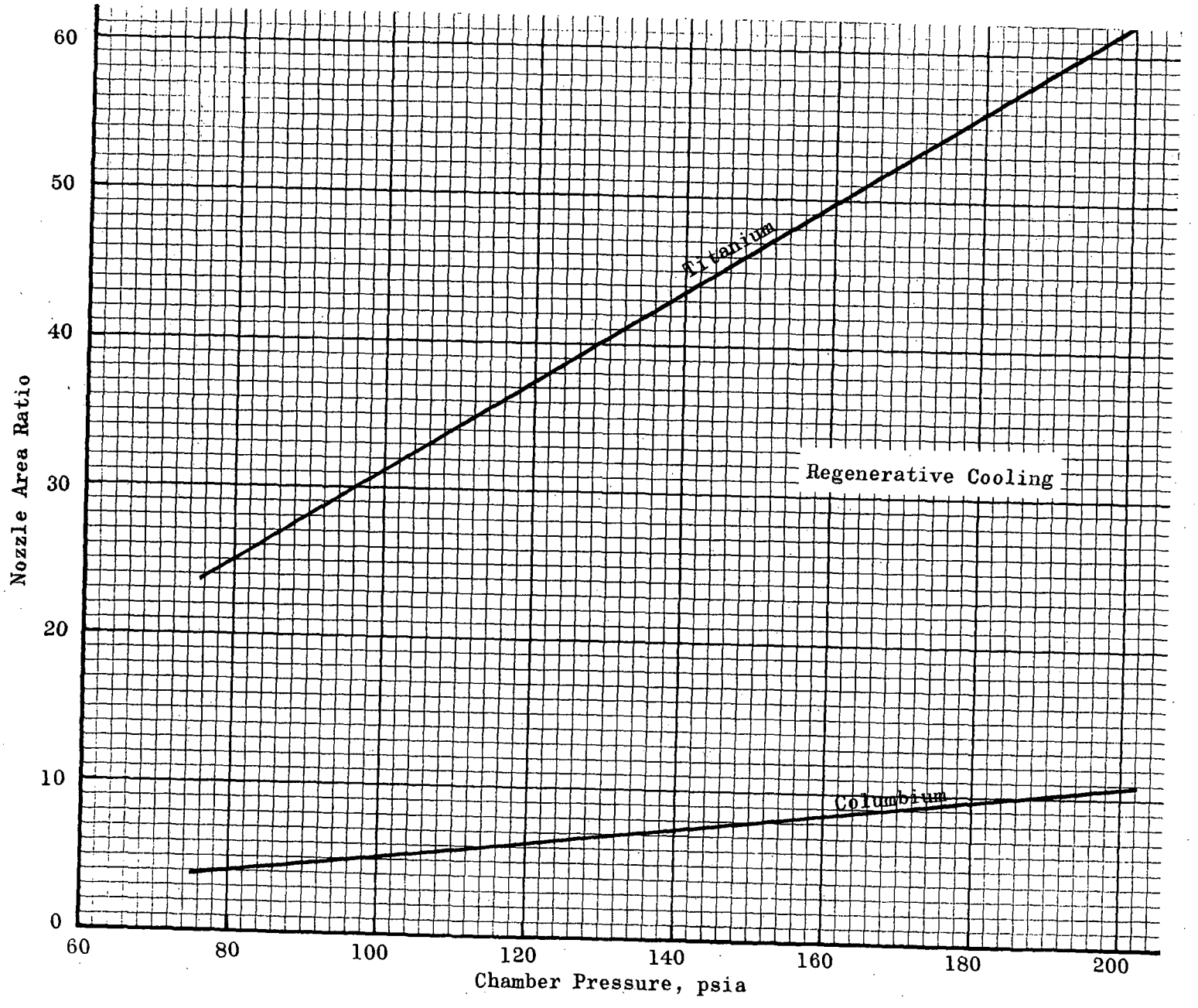


Figure 9. Effect of Pressure on Radiation Nozzle Attach Point

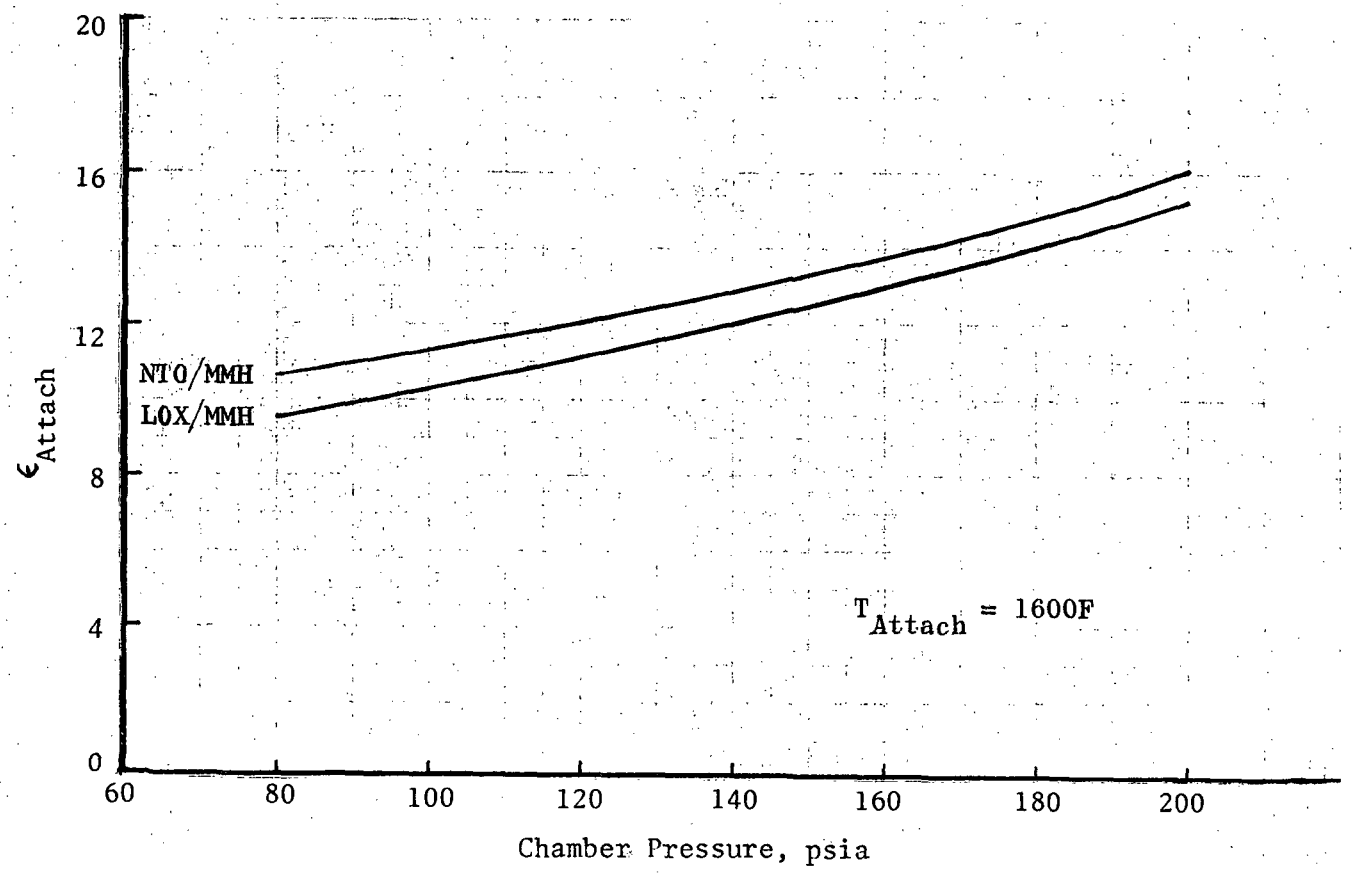


Figure 10. Titanium Radiation Nozzle Attach Point for Film-Cooled Engine

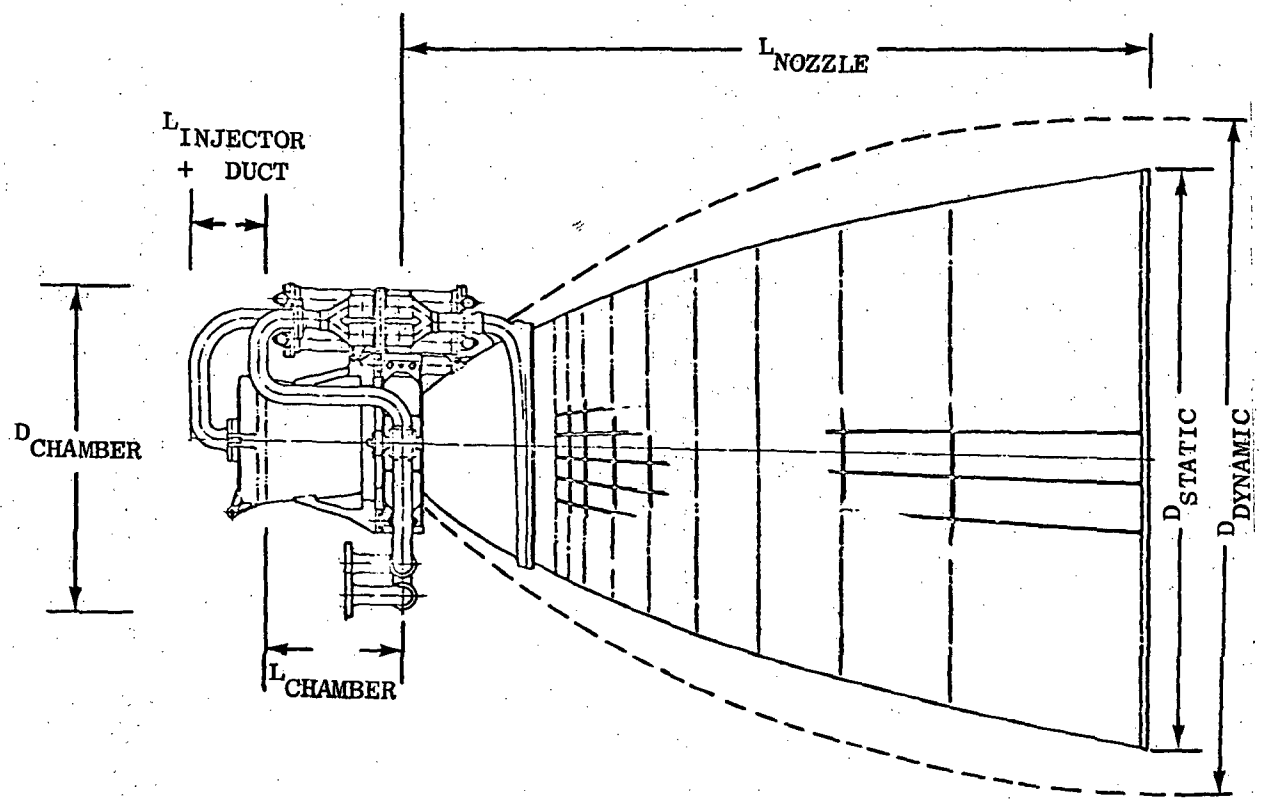


Figure 11. Engine Envelope

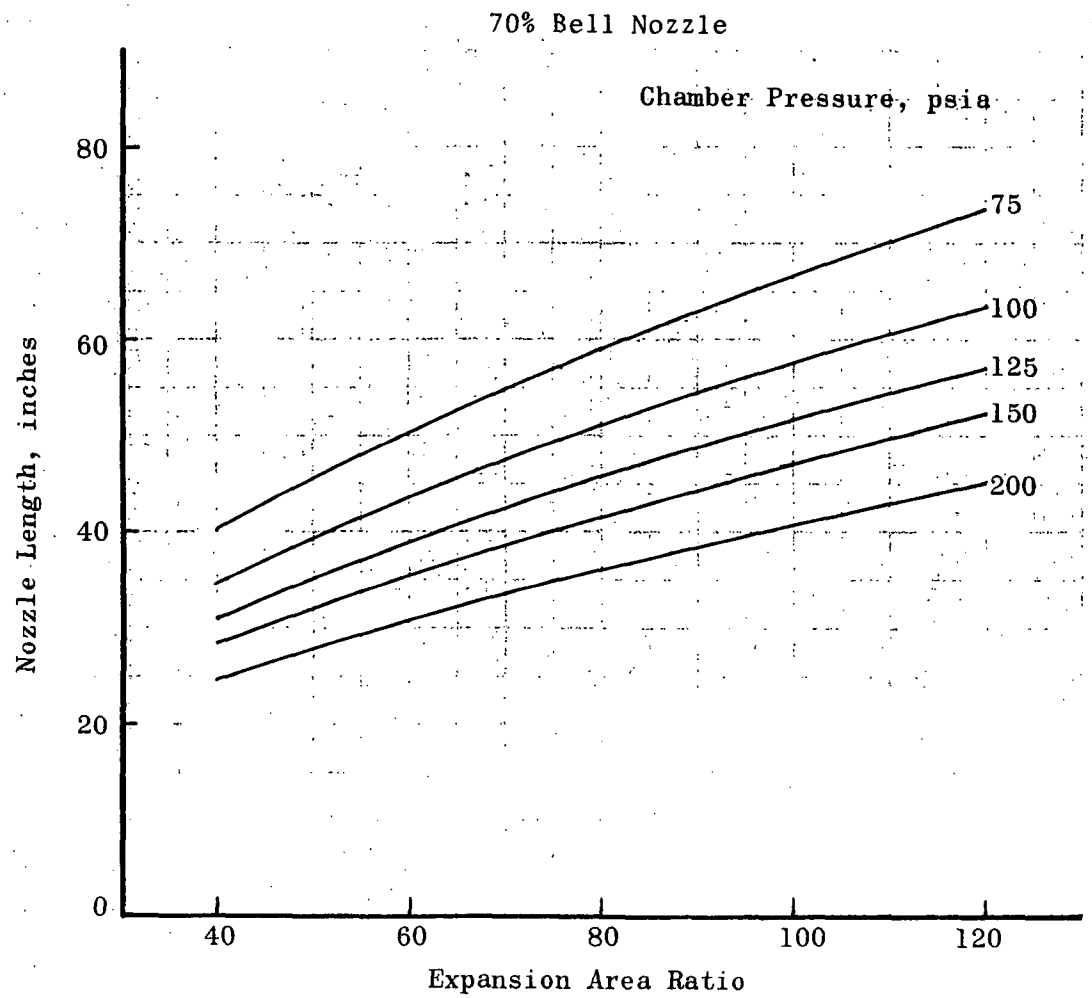


Figure 12. Nozzle Length at 3500 Pounds Thrust

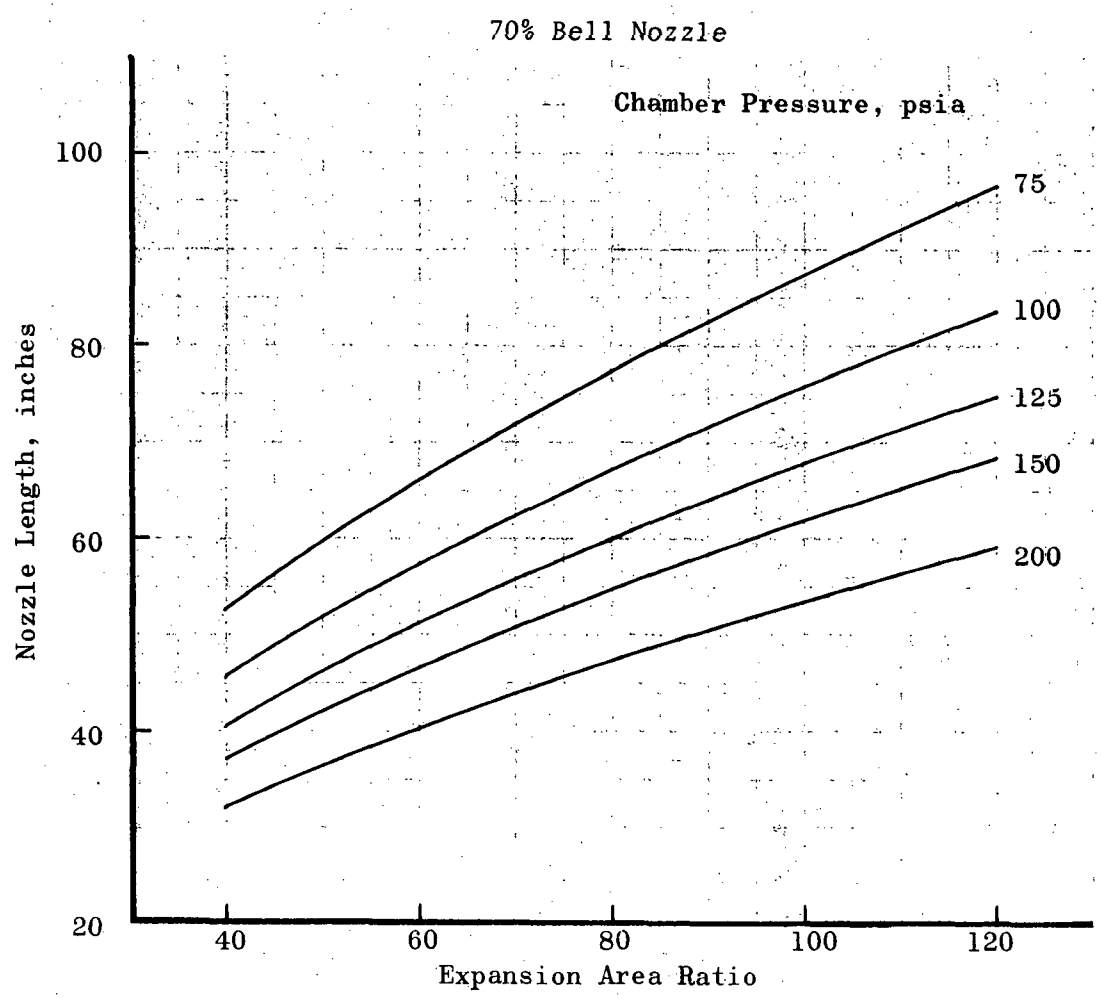


Figure 13. Nozzle Length at 6000 Pounds Thrust

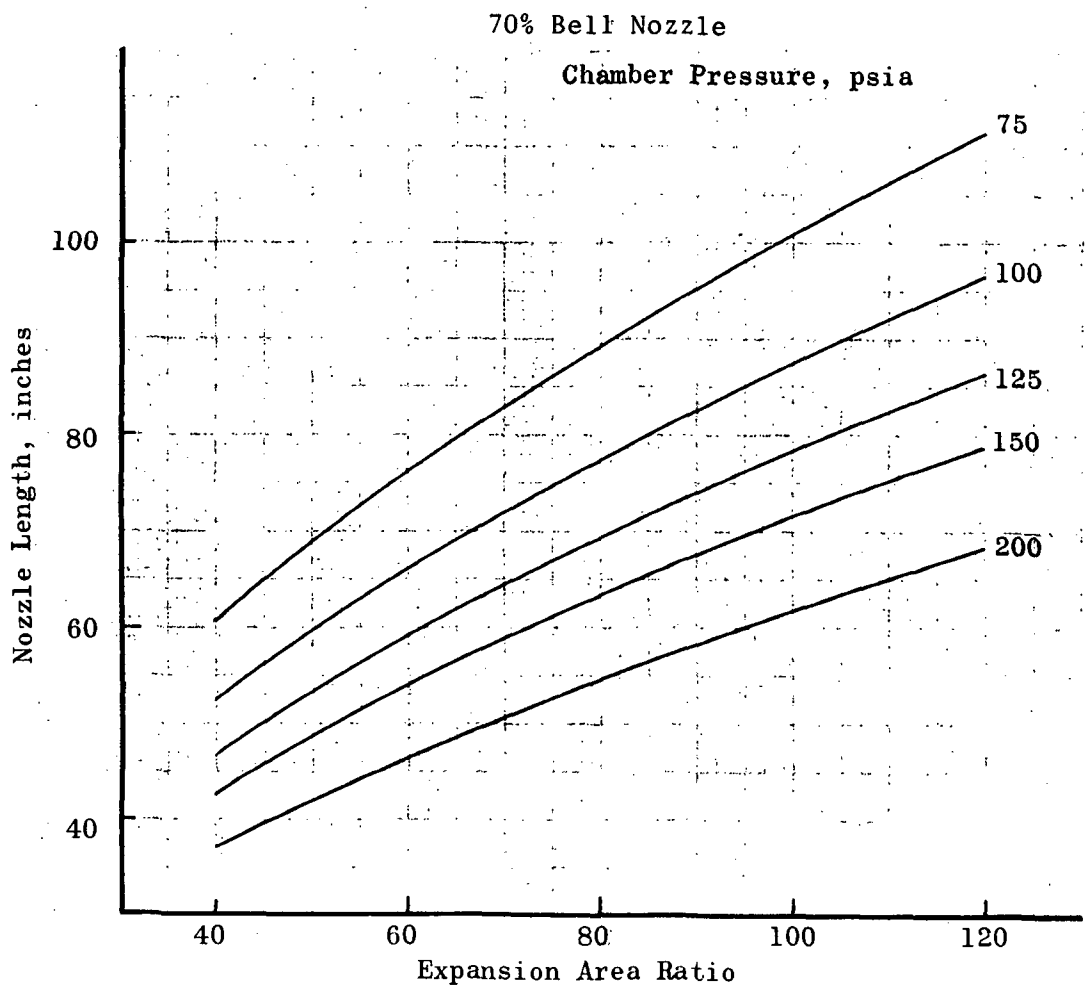


Figure 14. Nozzle Length at 8000 Pounds Thrust

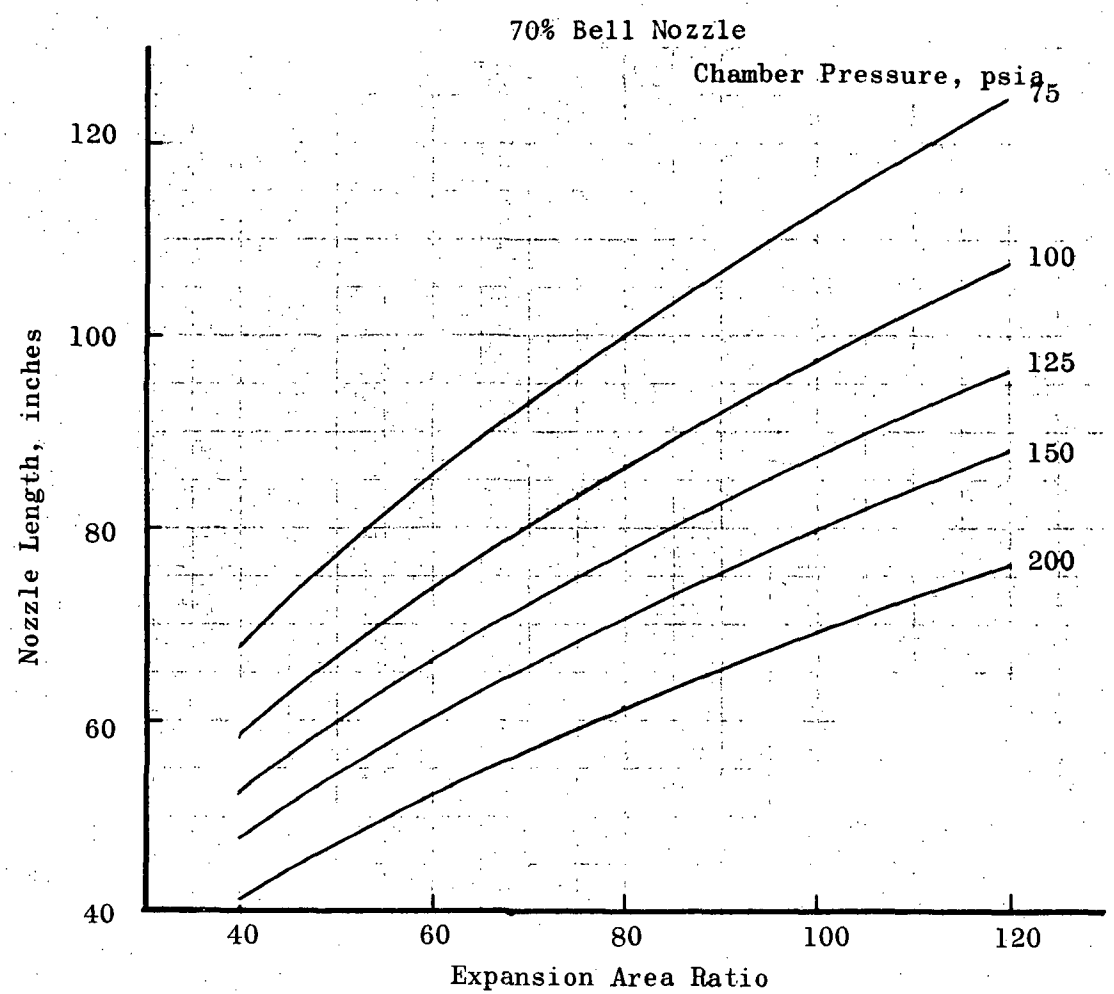
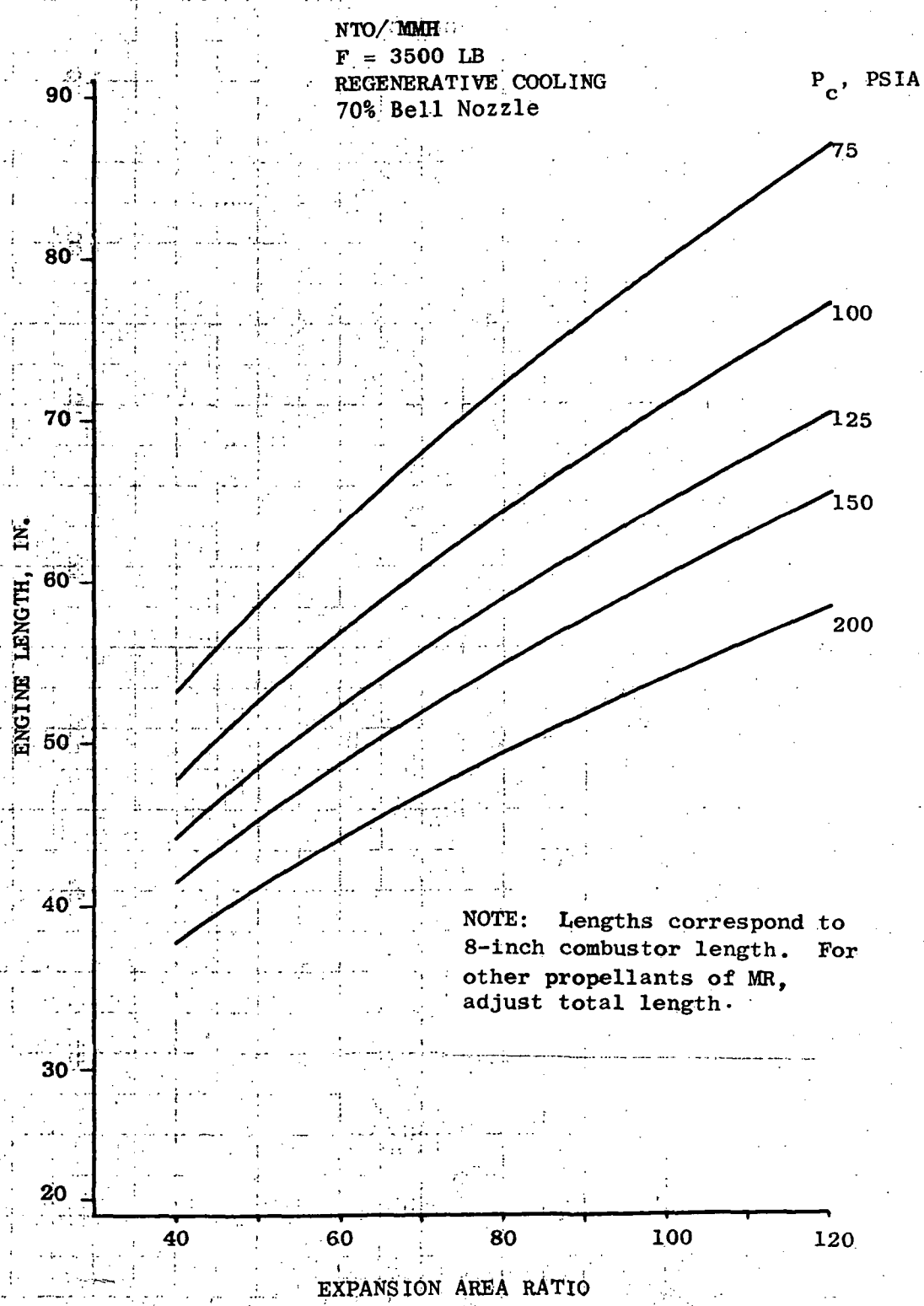


Figure 15. Nozzle Length at 10,000 Pounds Thrust

Figure 16

ENGINE LENGTH



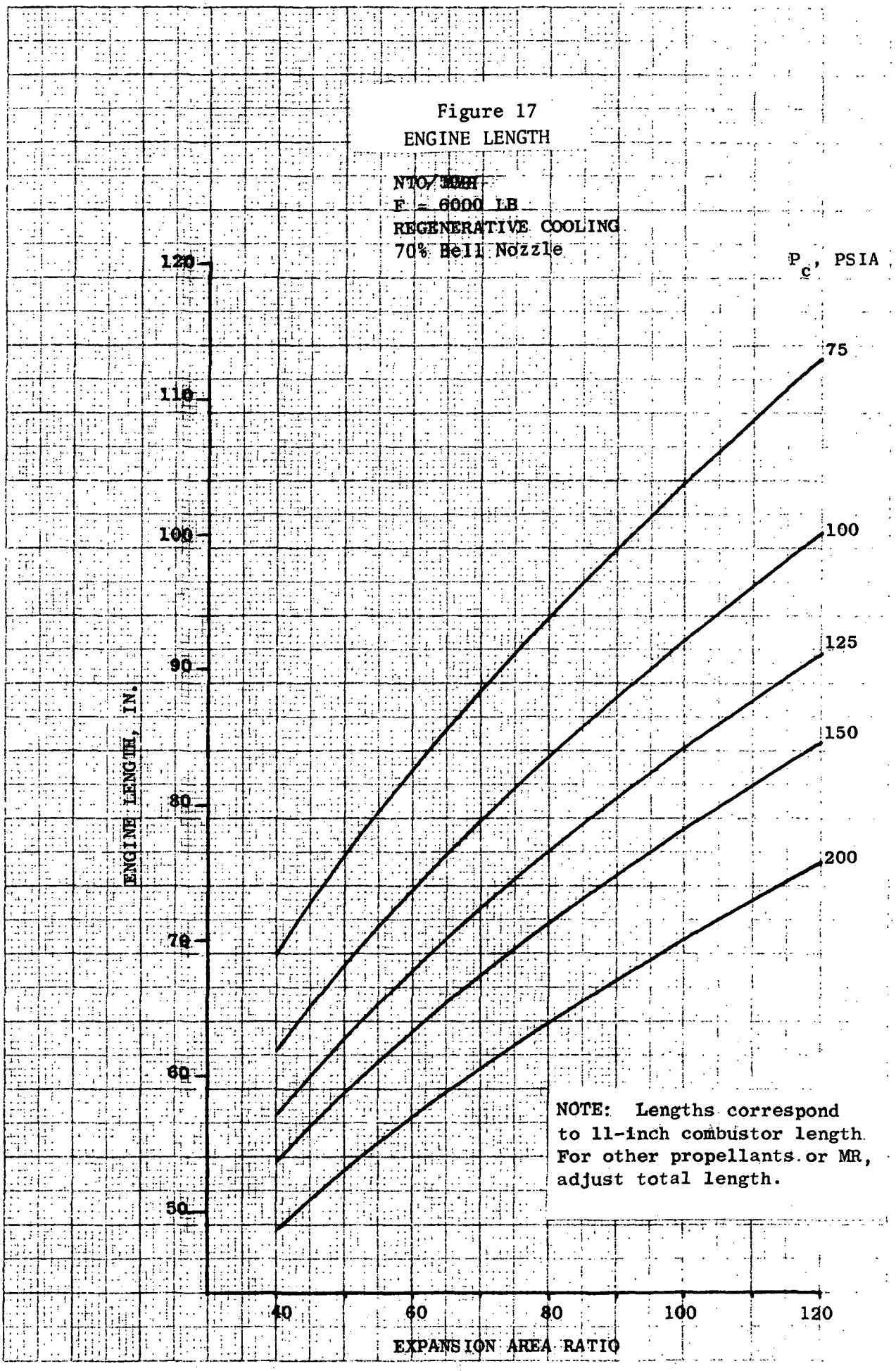


Figure 17
ENGINE LENGTH

Figure 18

ENGINE LENGTH

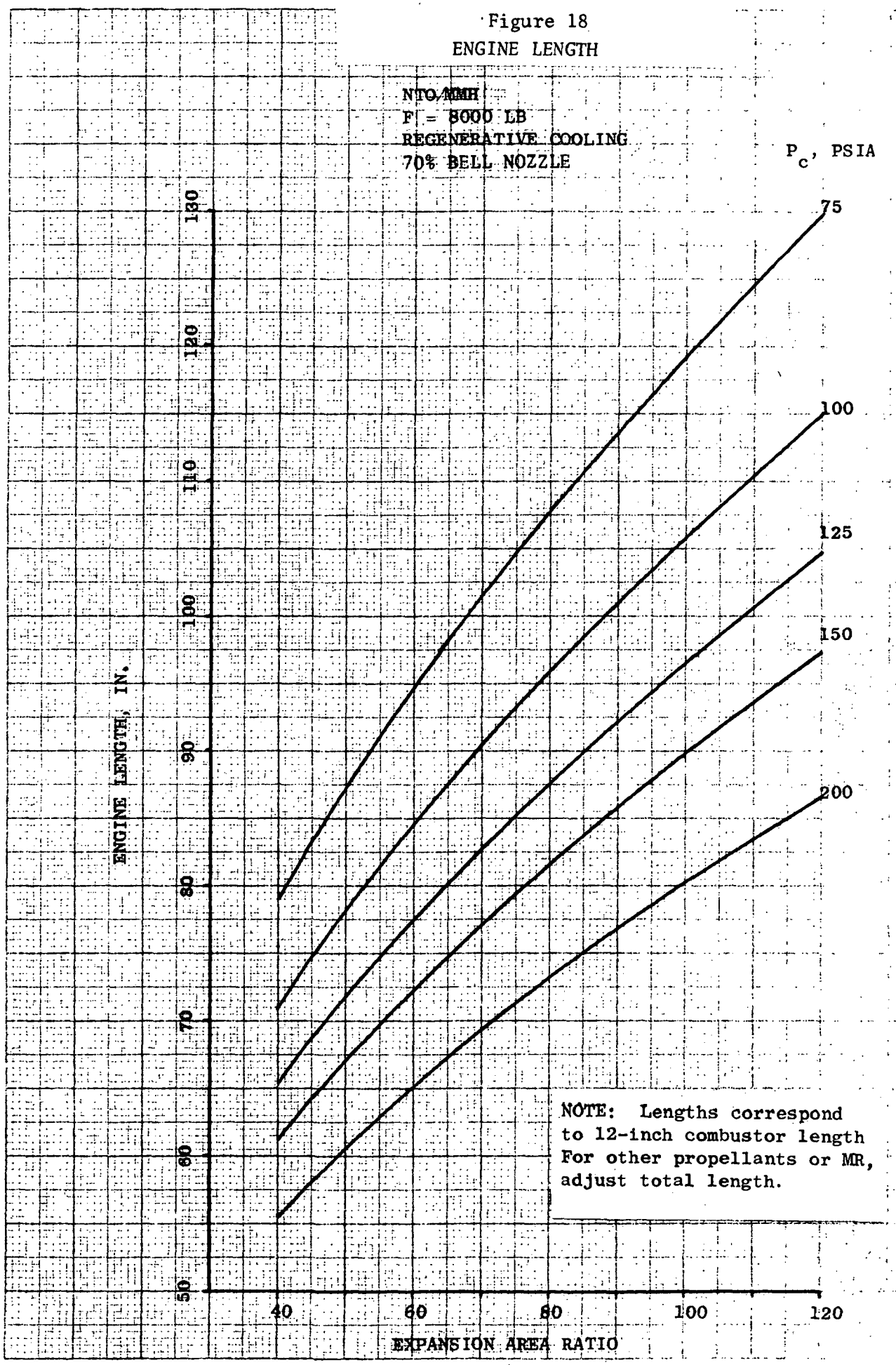
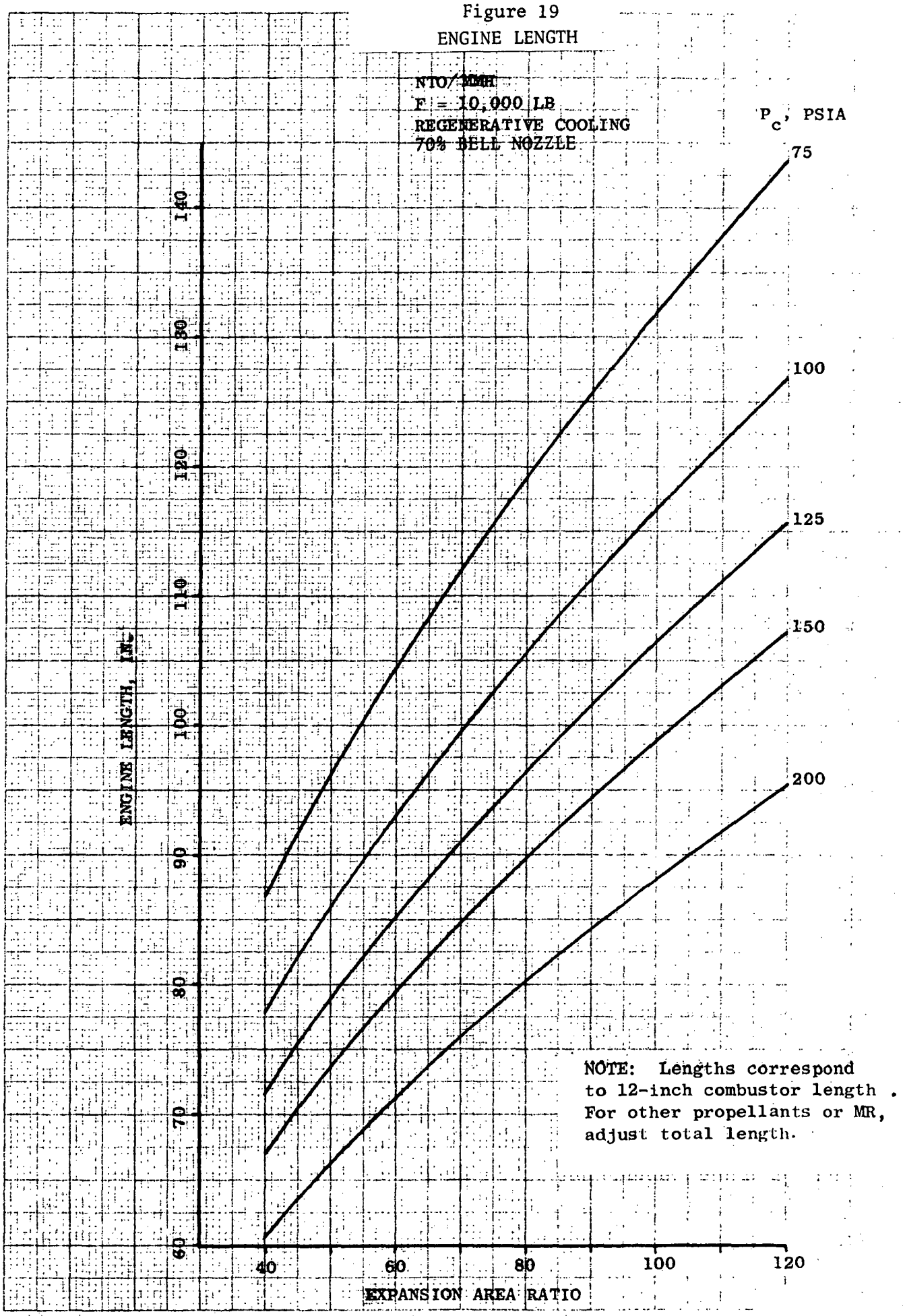


Figure 19

ENGINE LENGTH



NTO/MMH
Thrust = 6000 pounds
Chamber Pressure = 125 psia
Exit Diameter = 50 inches
Mixture Ratio = 1.65
Injector-to-throat
Length = 11 inches
Thrust Chamber Length =
68 inches

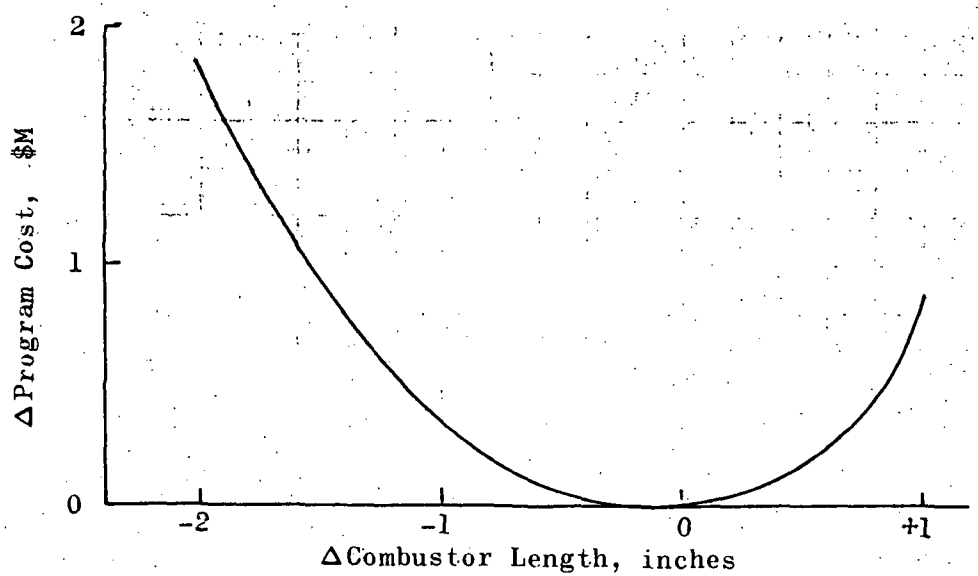


Figure 20a. Optimization of Combustor Length at Constant Engine Length

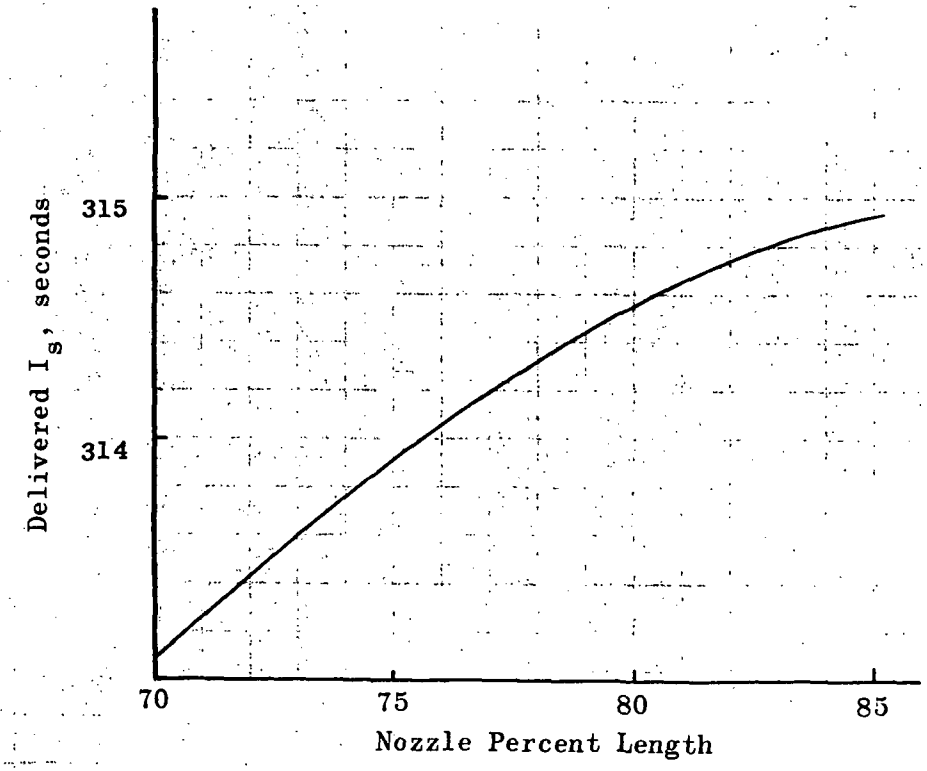


Figure 20b. Nozzle Contour Effect

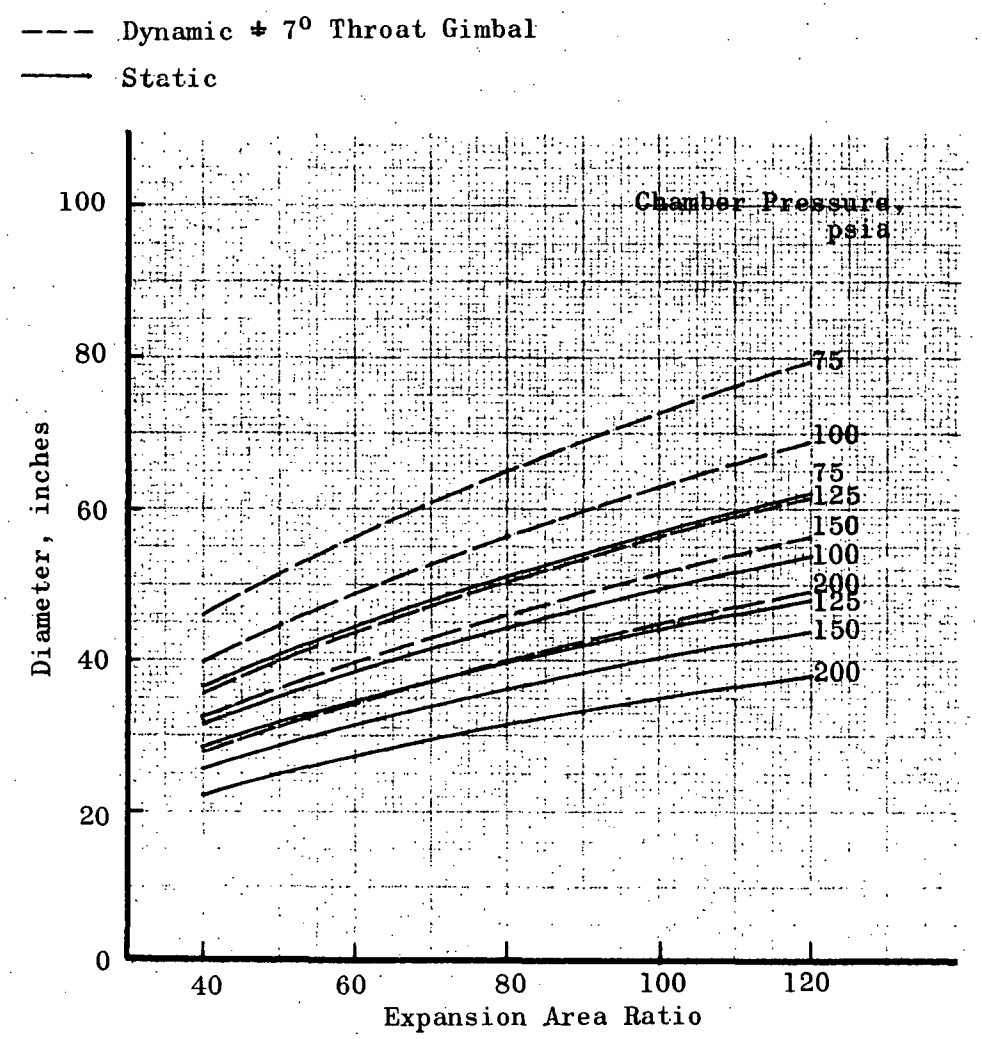


Figure 21. Maximum Diameter at 3500 Pounds Thrust

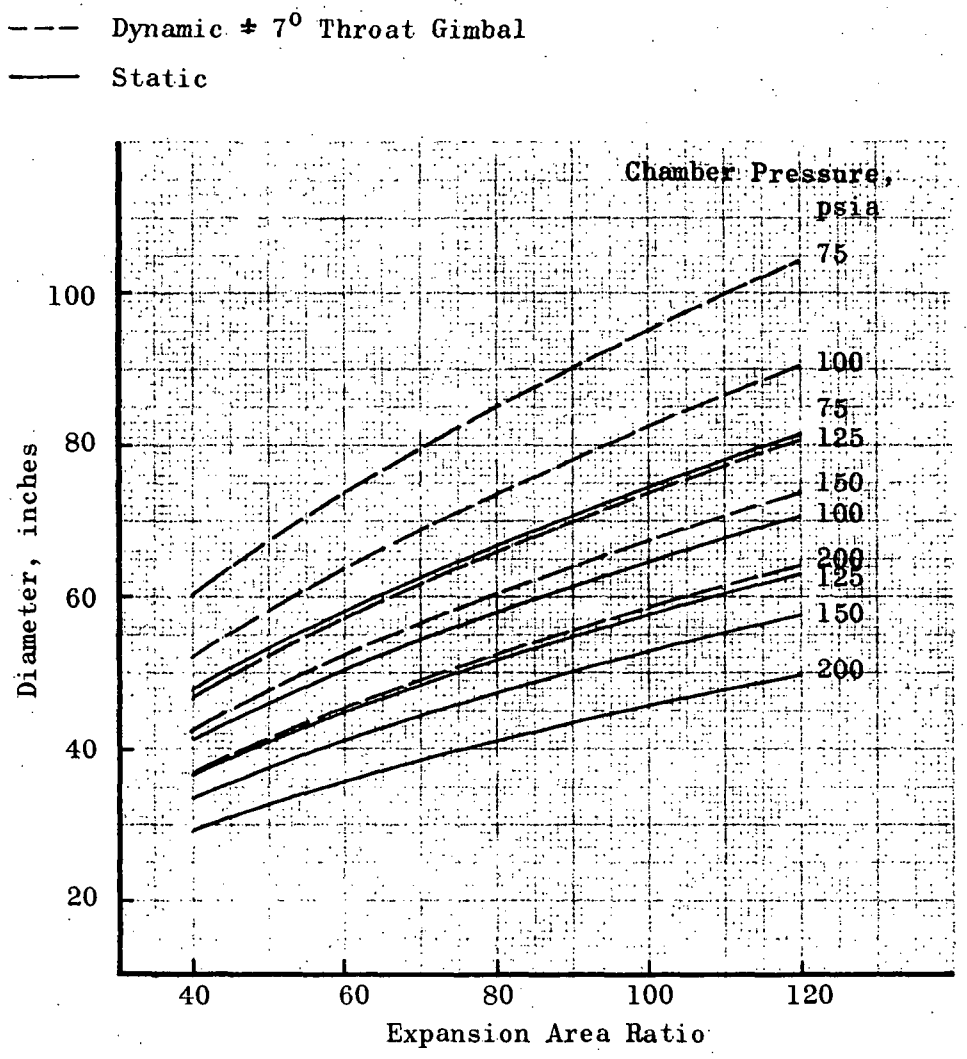


Figure 22. Maximum Diameter at 6000 Pounds Thrust

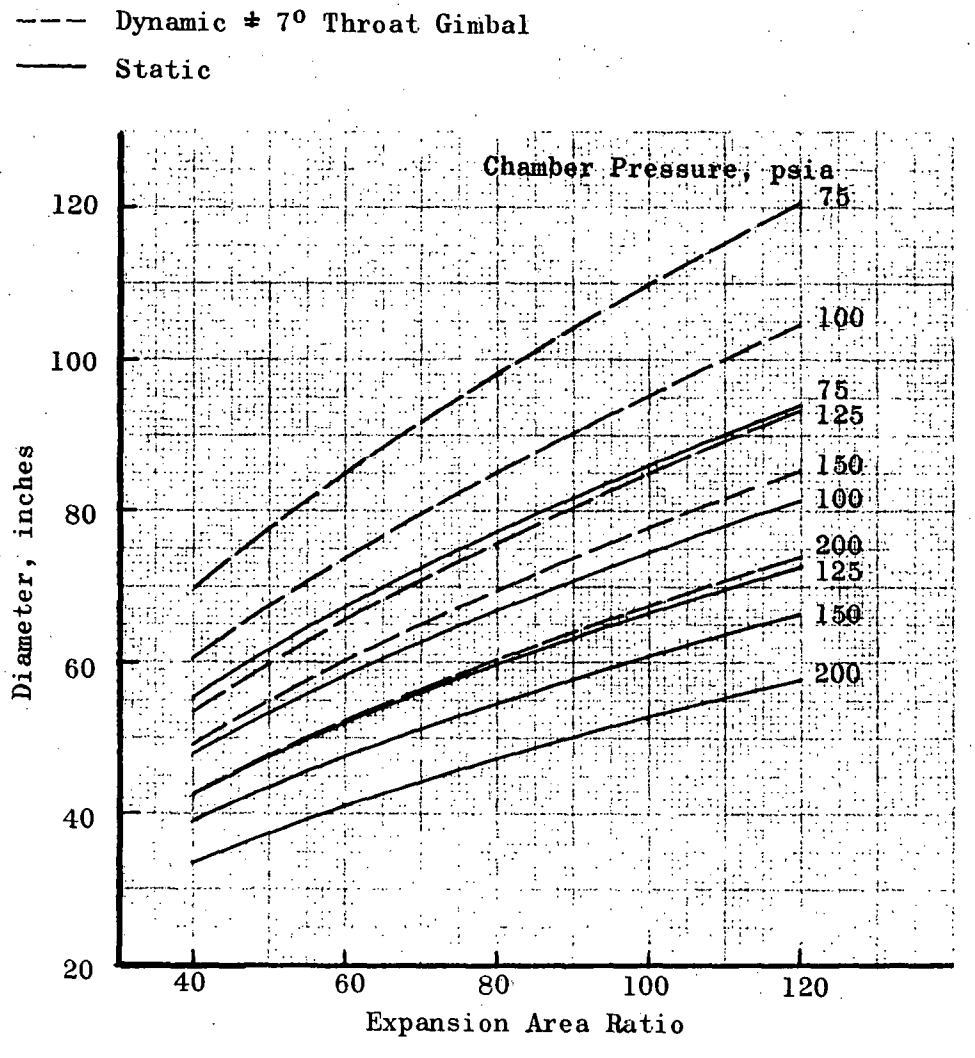


Figure 23. Maximum Diameter at 8000 Pounds Thrust

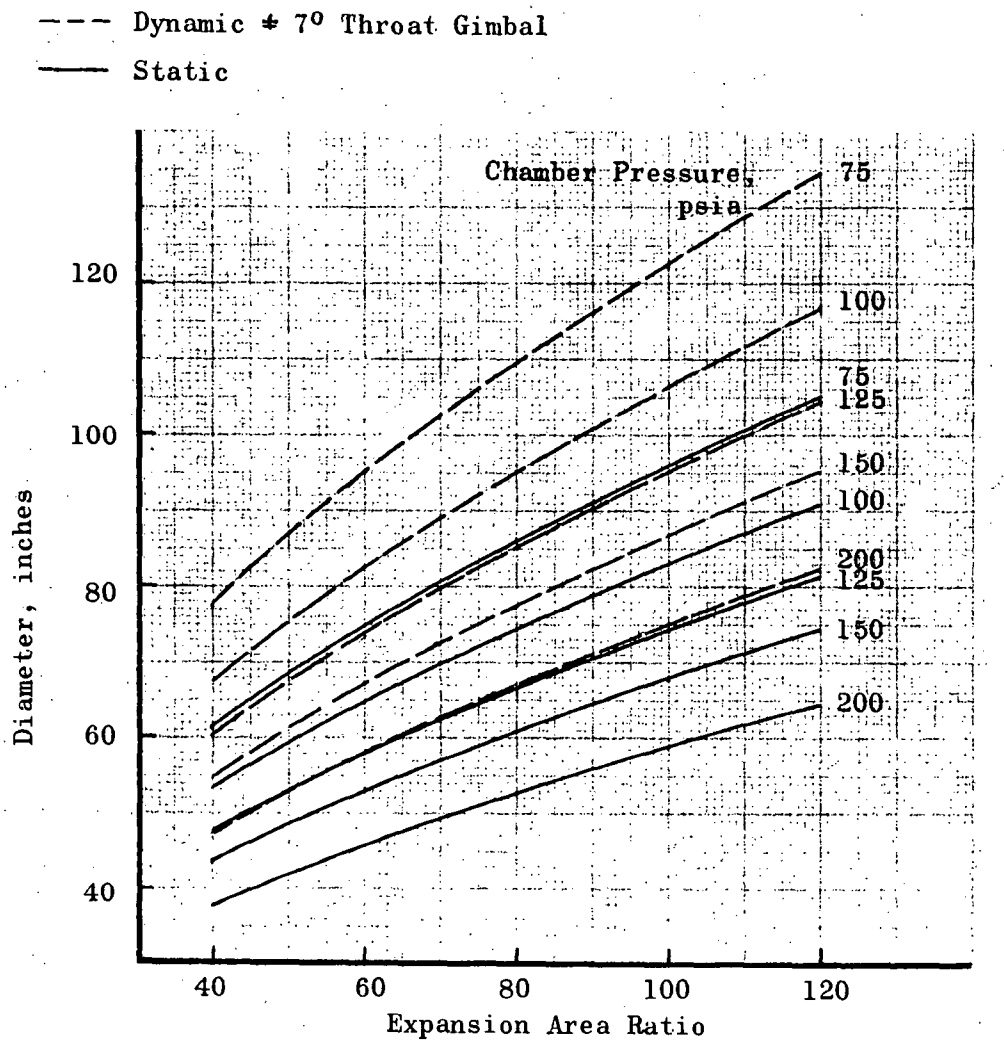


Figure 24. Maximum Diameter at 10,000 Pounds Thrust

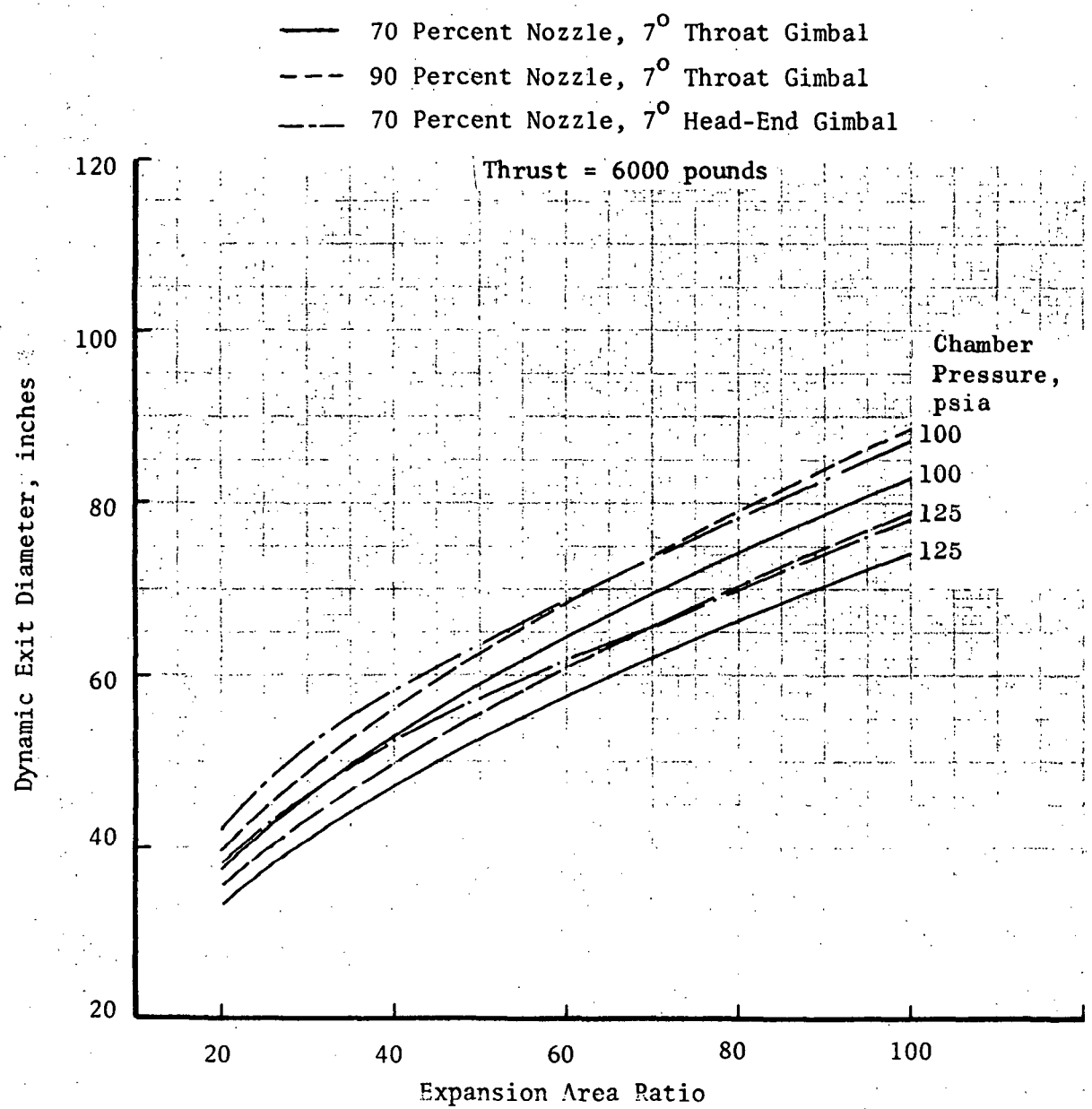


Figure 25. Design Effects on Dynamic Diameter

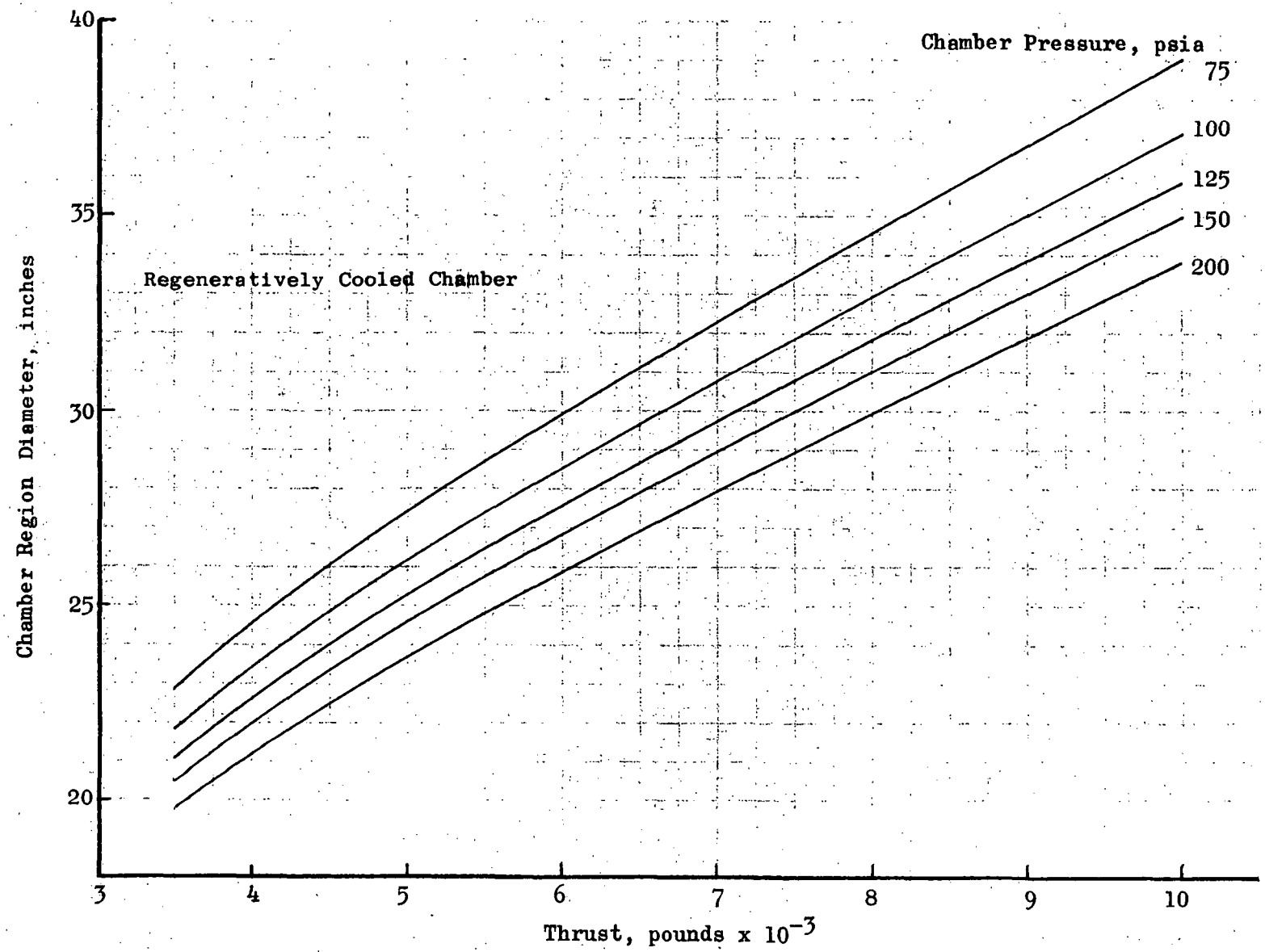


Figure 26. Chamber Region Diameter

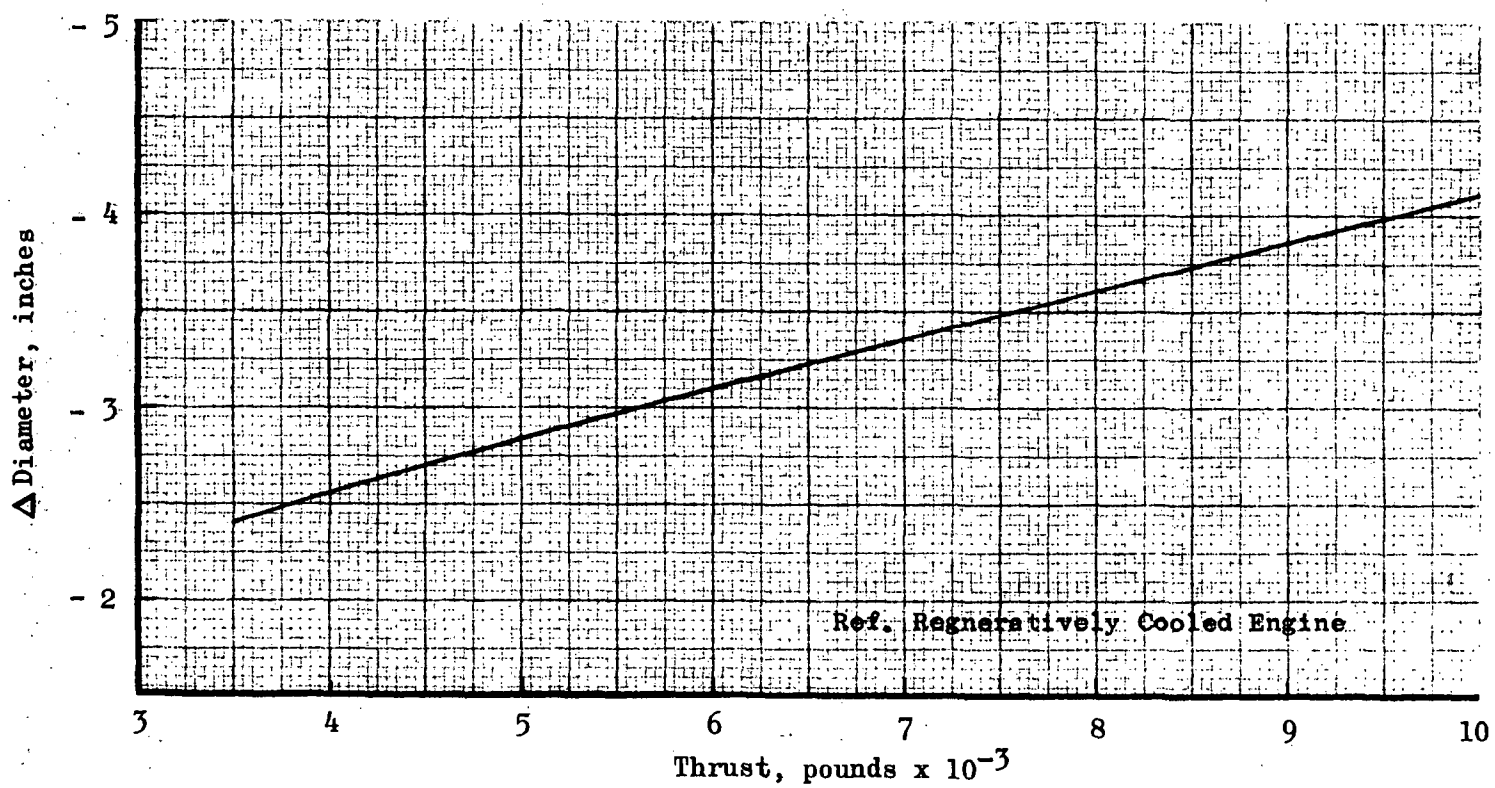


Figure 27. Change in Chamber Region Diameter for Dump/Film Cooled Engine

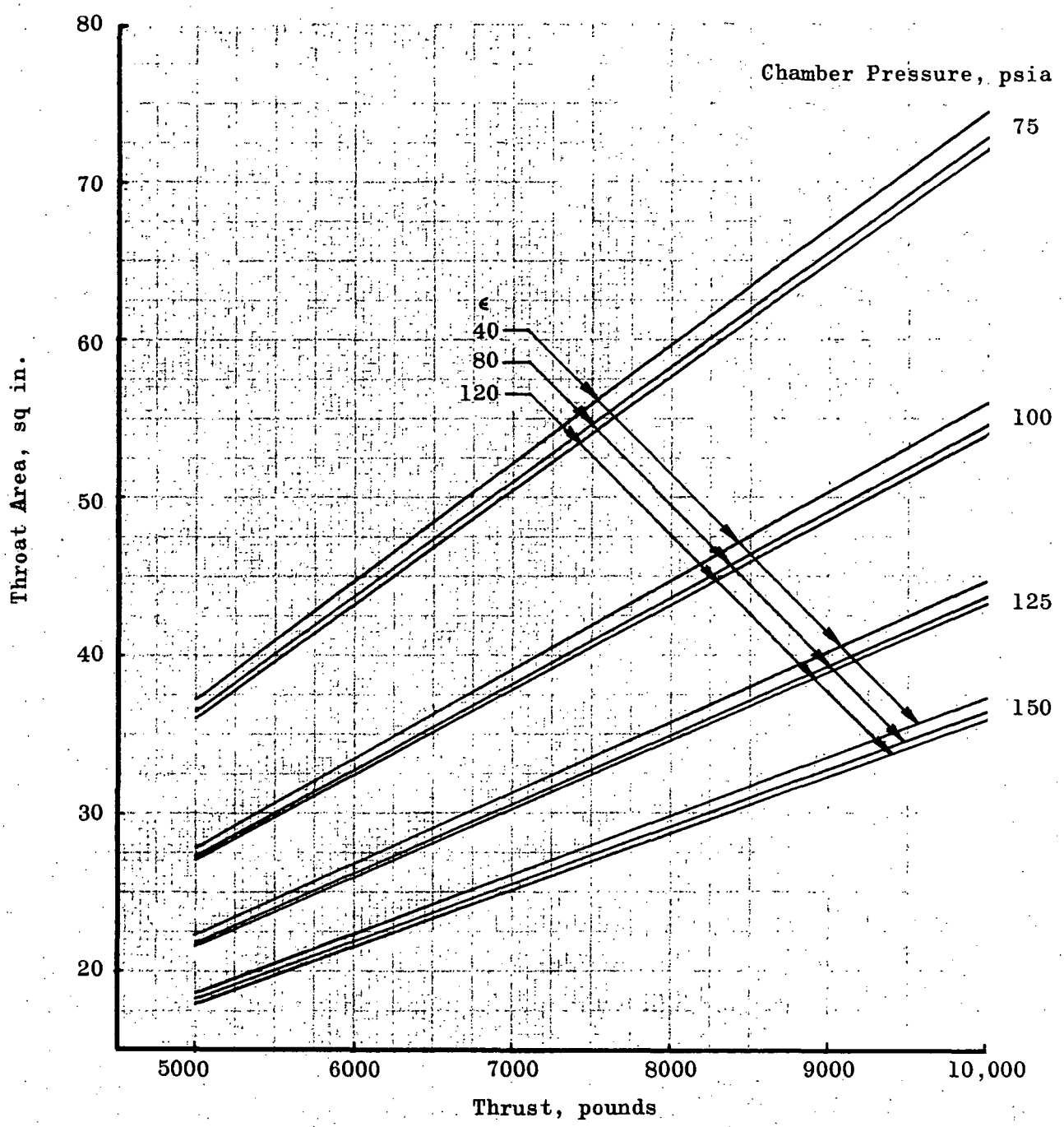


Figure 28. OME Engine Throat Area

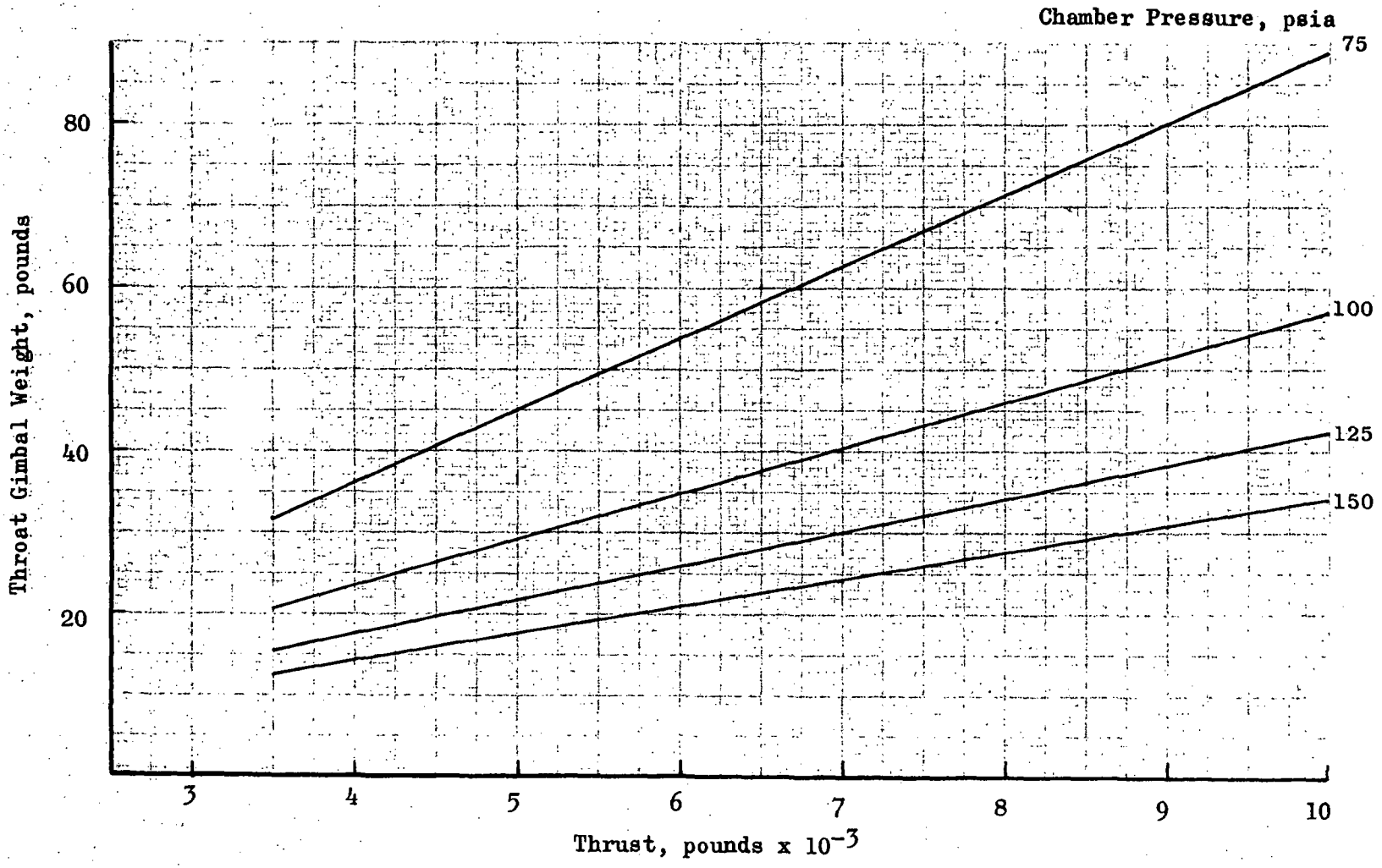


Figure 29. Throat Gimbal Weight

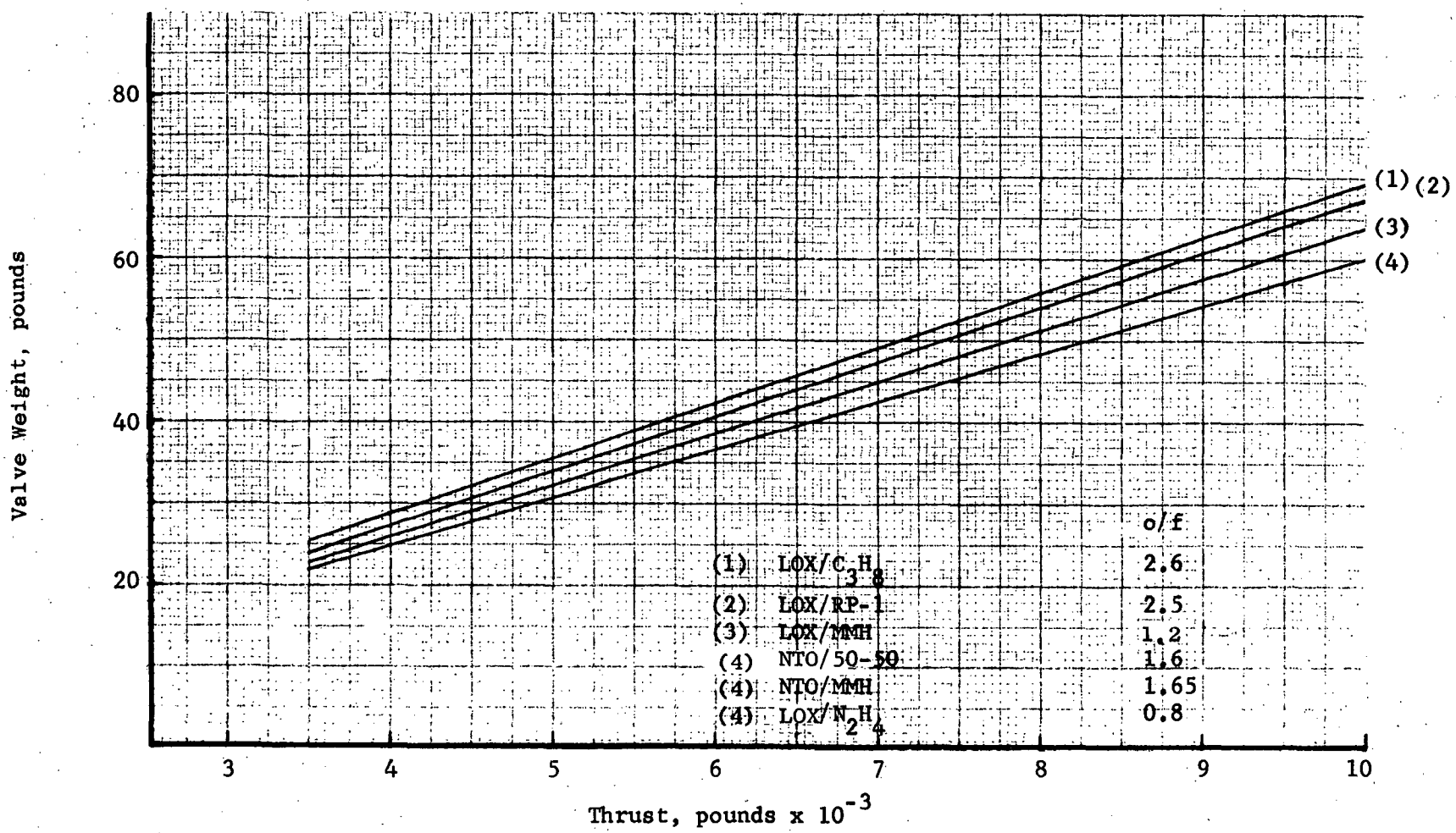


Figure 30. Parallel/Series Ball Valve Weight

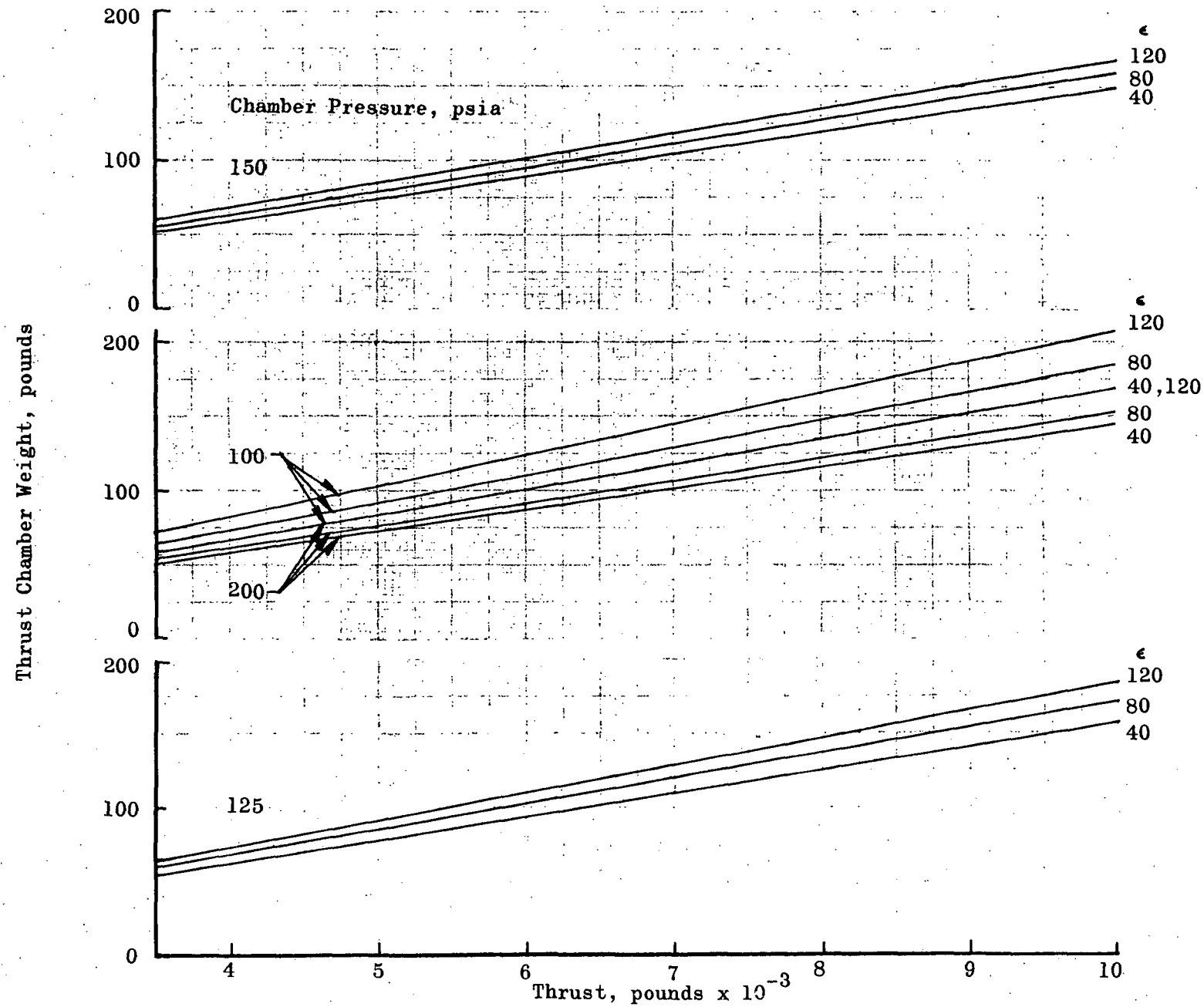


Figure 31. Regeneratively Cooled Thrust Chamber Assembly Weight, NTO/MMH and NTO/50-50

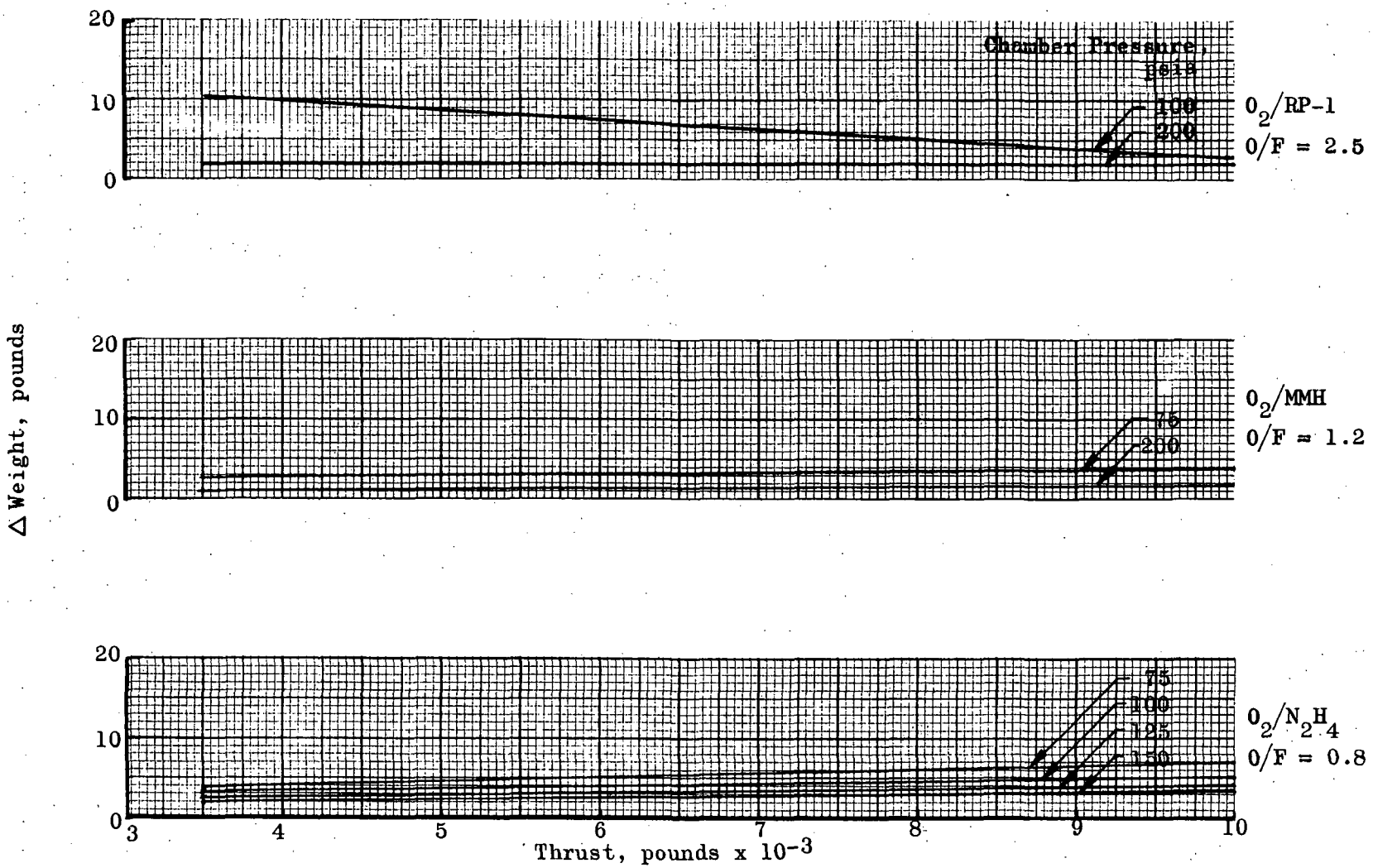


Figure 32. O_2/RP-1, O_2/MMH, O_2/N_2H_4 Regenerative Cooled Thrust Chamber Assembly Weight Increase Relative to NTO/MMH Design

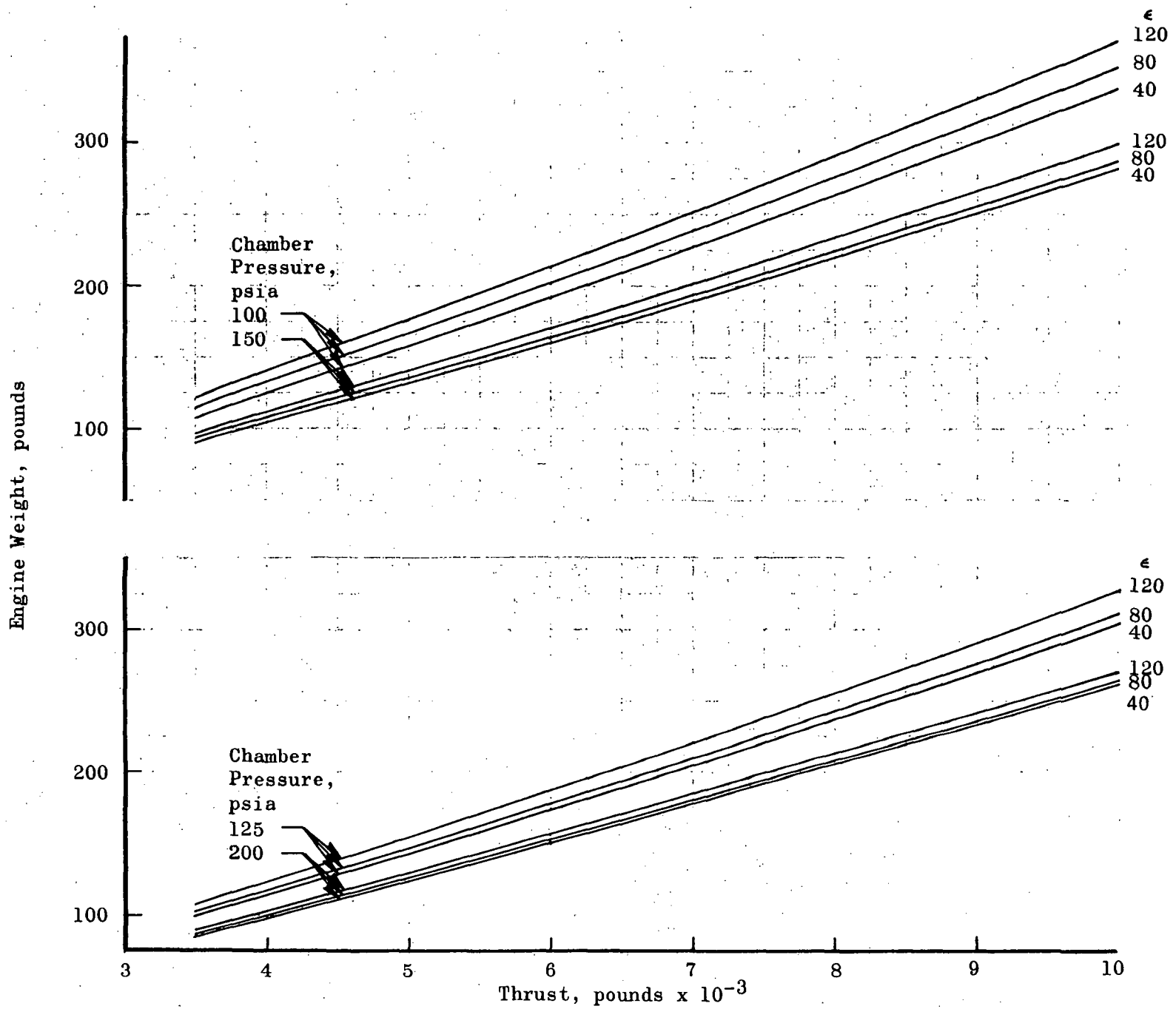


Figure 33. Regeneratively Cooled Engine Weight NTO/MMH and NTO/50-50

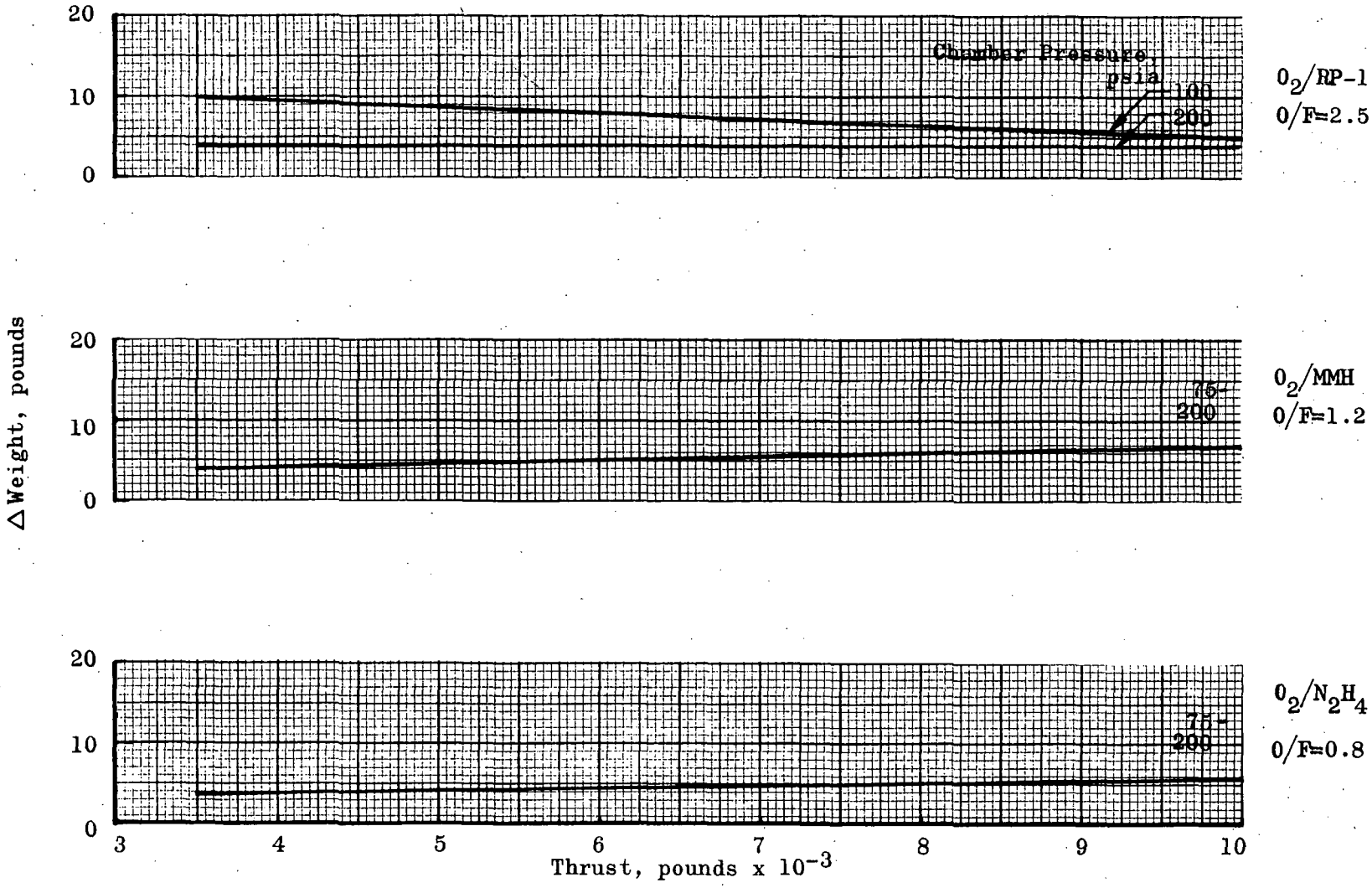


Figure 34. O_2/RP-1, O_2/MMH, O_2/N_2H_4 Regenerative Cooled Engine Weight Increase Relative to NTO/MMH Design

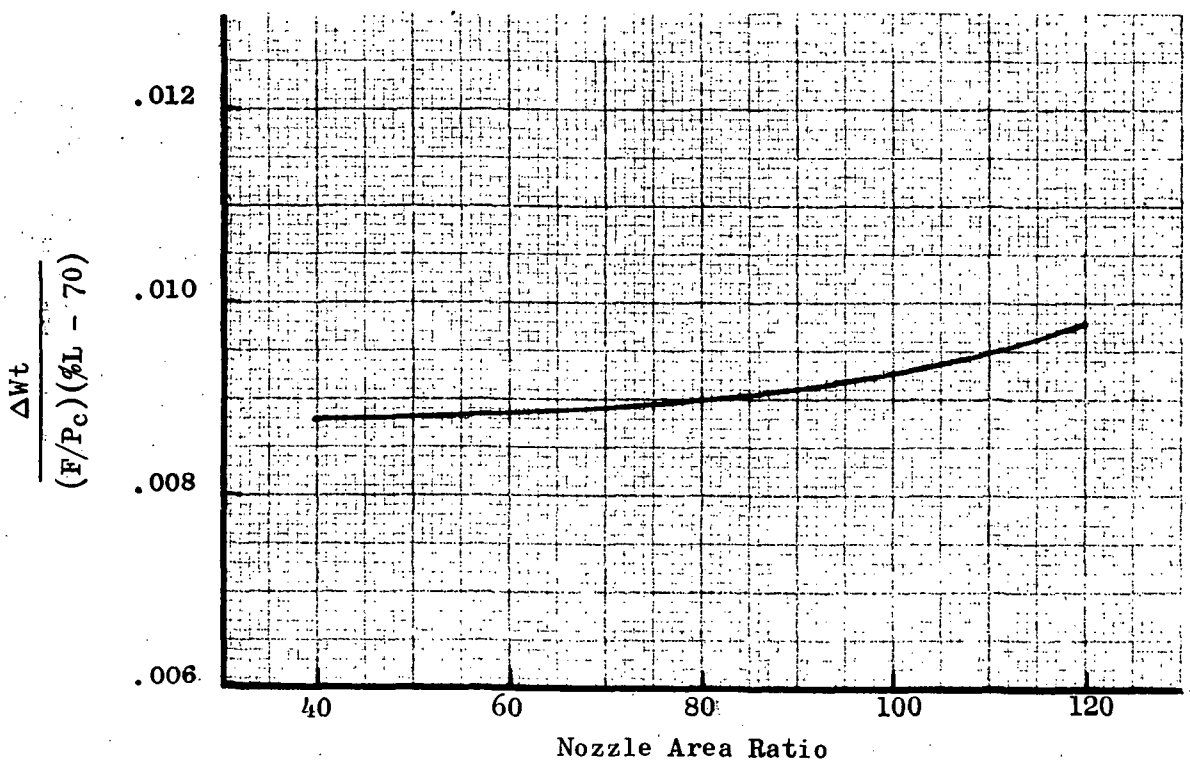


Figure 35. Effect of Nozzle Contour on (Regeneratively Cooled) Engine Weight

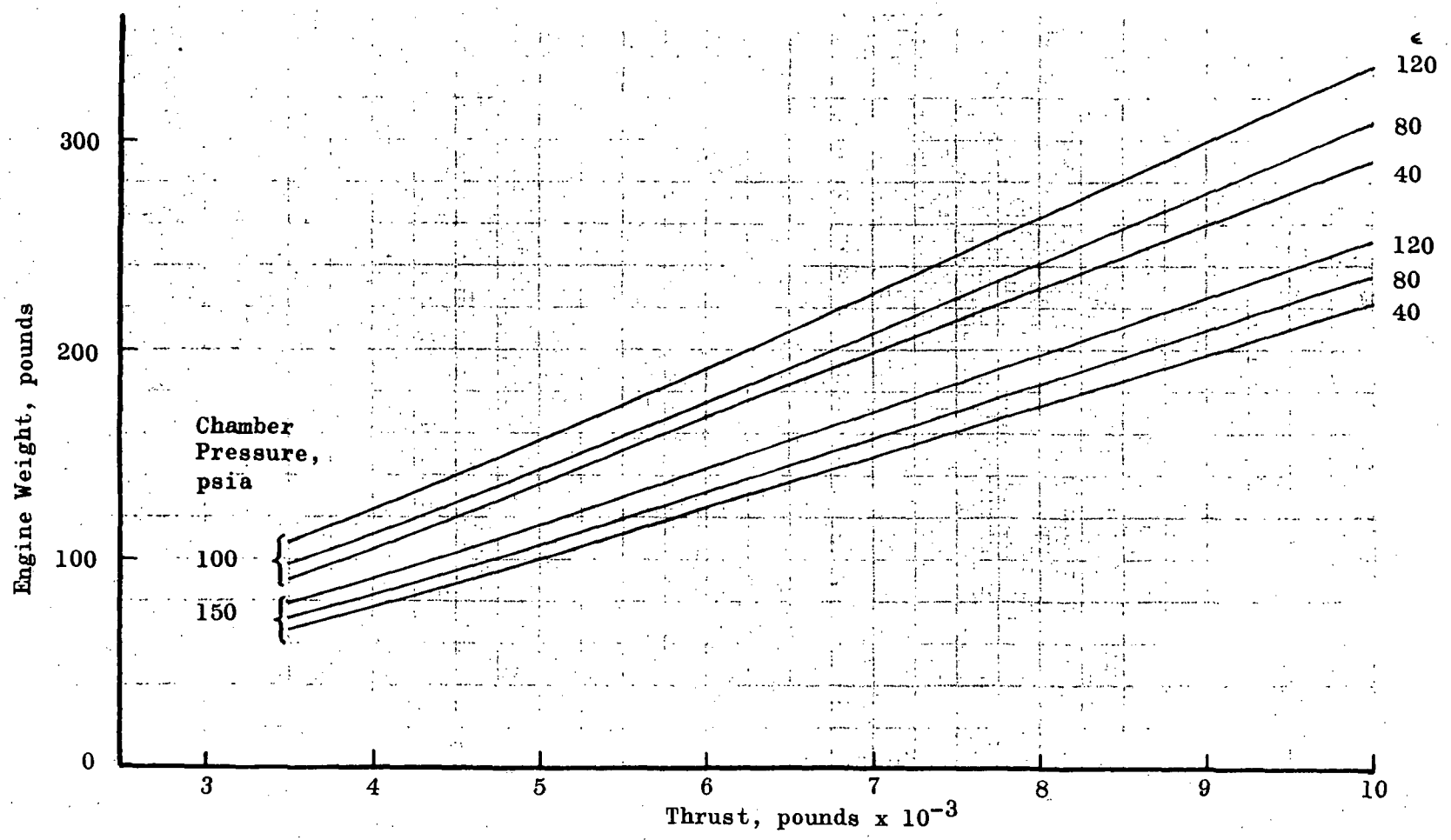


Figure 36. Dump/Film Cooled Engine Weights, NTO/MMH and NTO/50-50

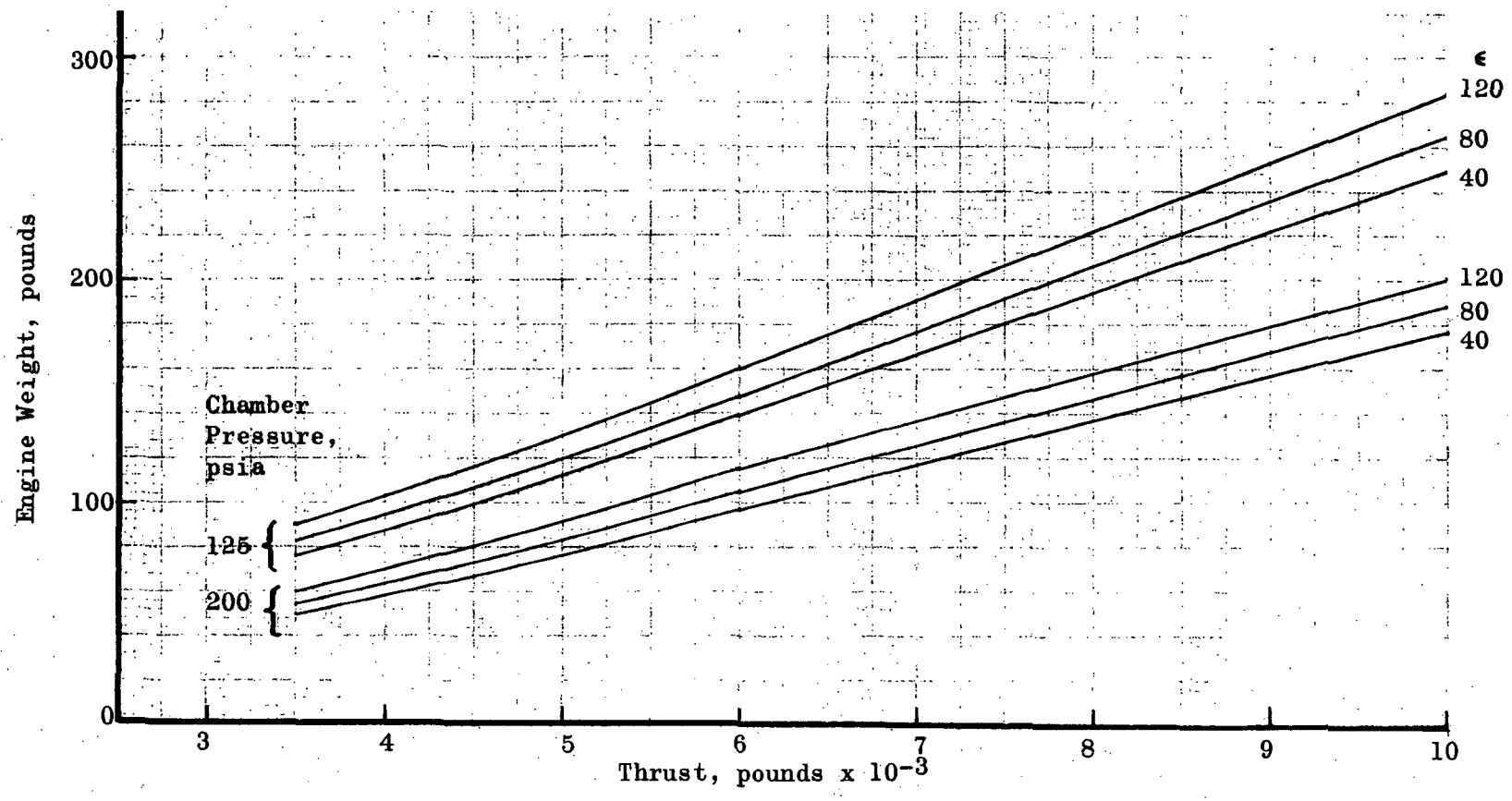


Figure 37. Dump/Film Cooled Engine Weights, NTO/MMH and NTO/50-50

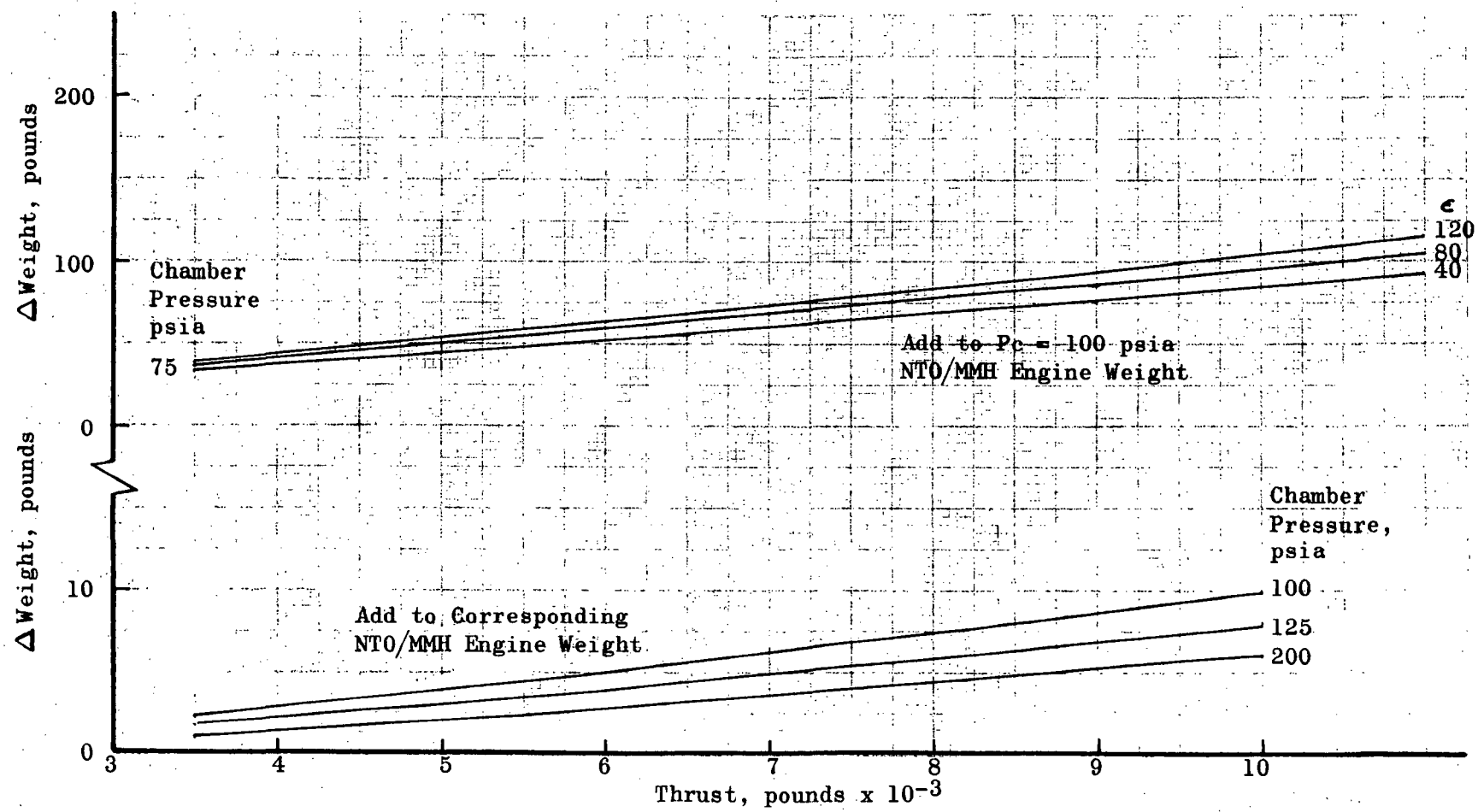


Figure 38. O_2/MMH and O_2/50-50 Dump/Film Cooled Engine Weight Increase Relative to NTO/MMH Design

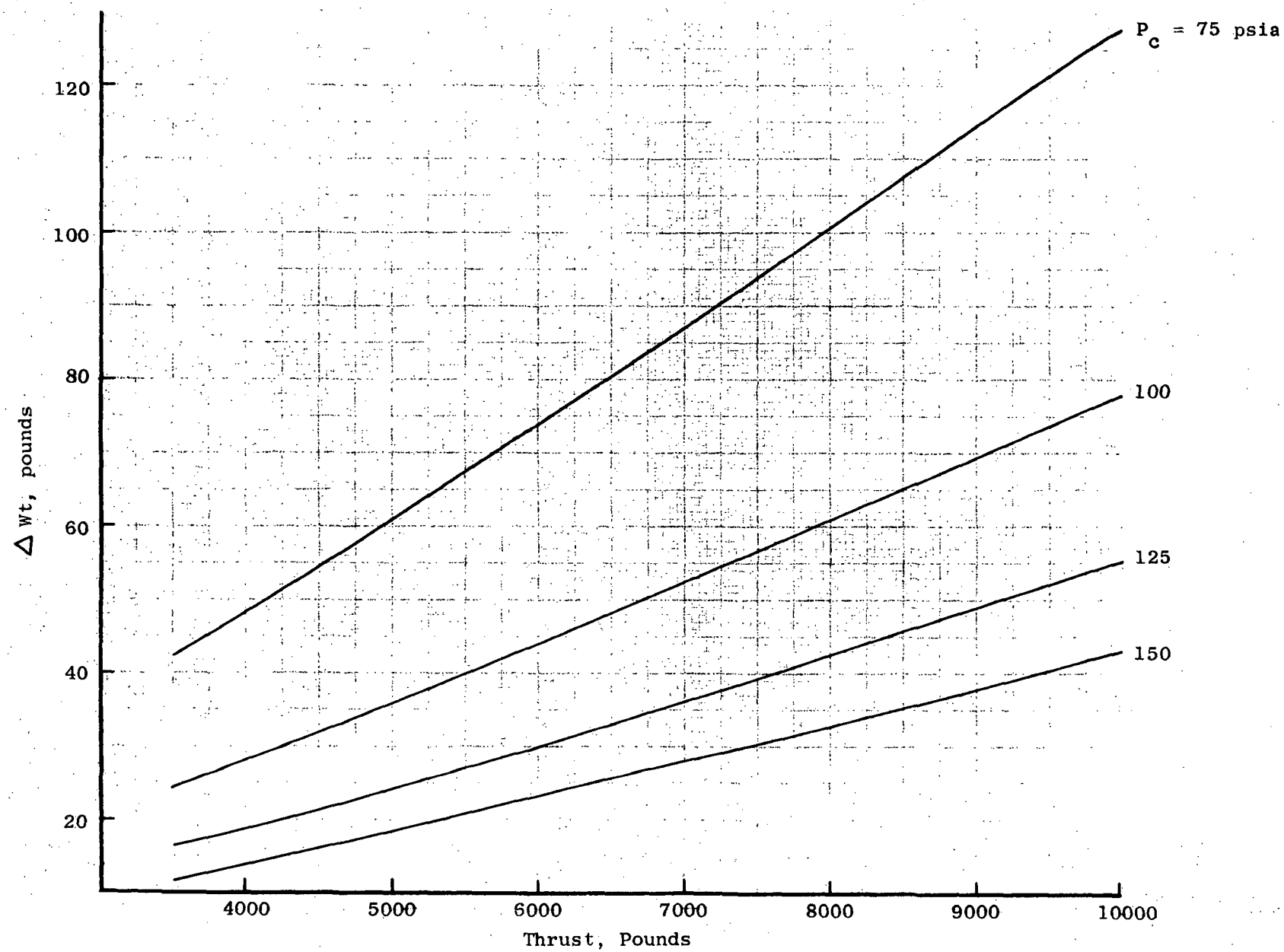


Figure 39. Engine Weight Savings for Non-Gimbaled Configuration

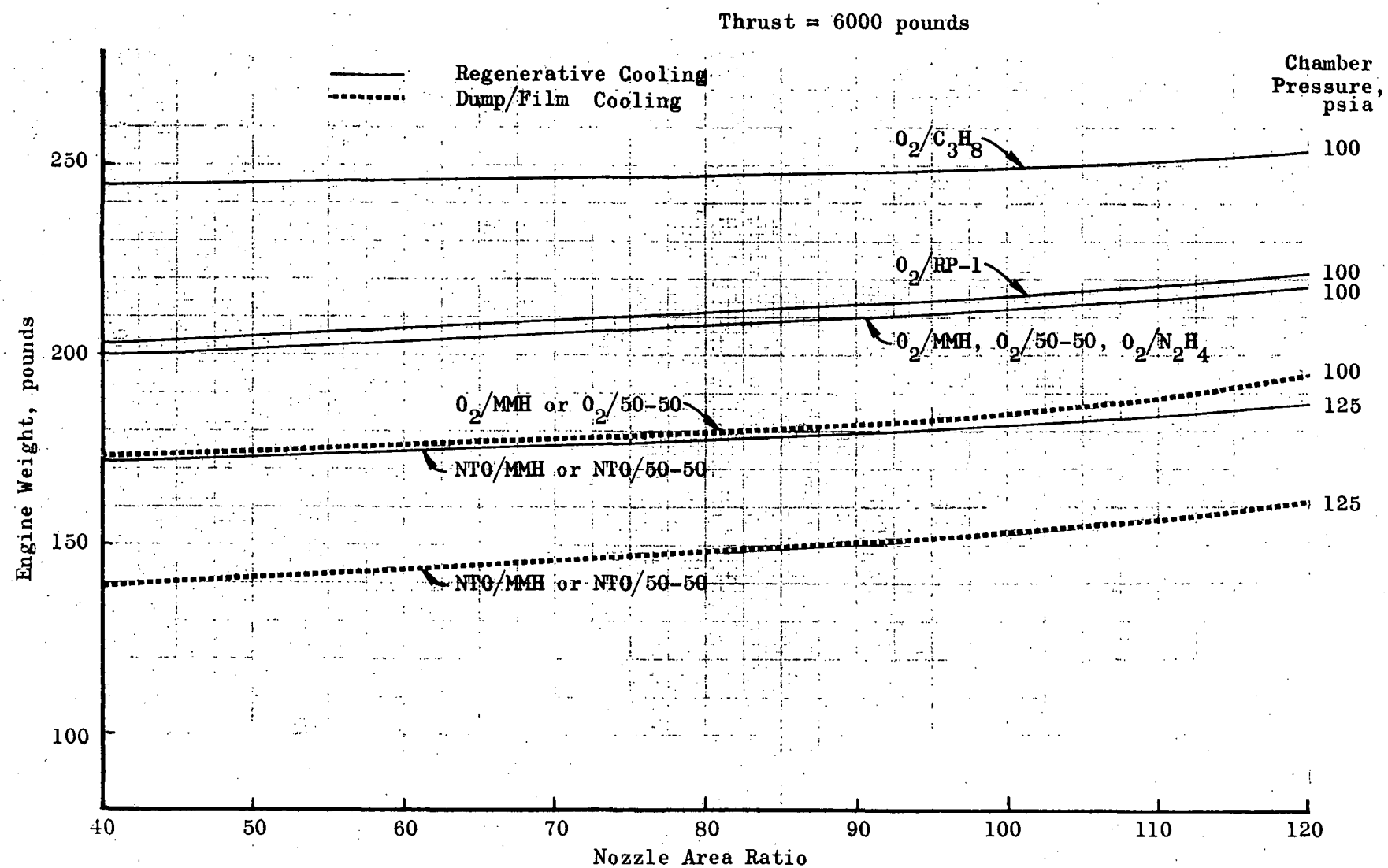
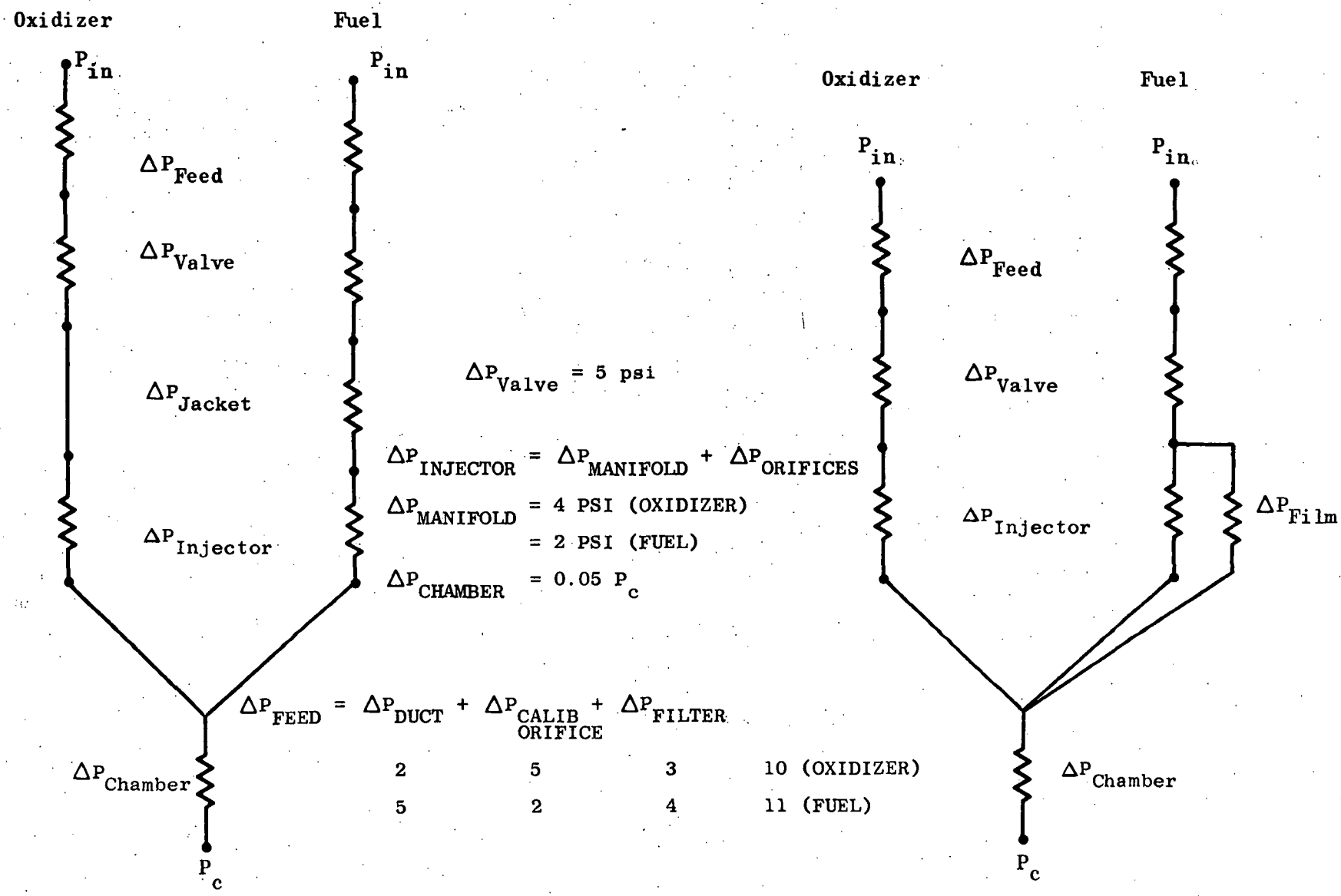


Figure 40. Engine Weight Comparisons

Figure 41
ENGINE INLET PRESSURE



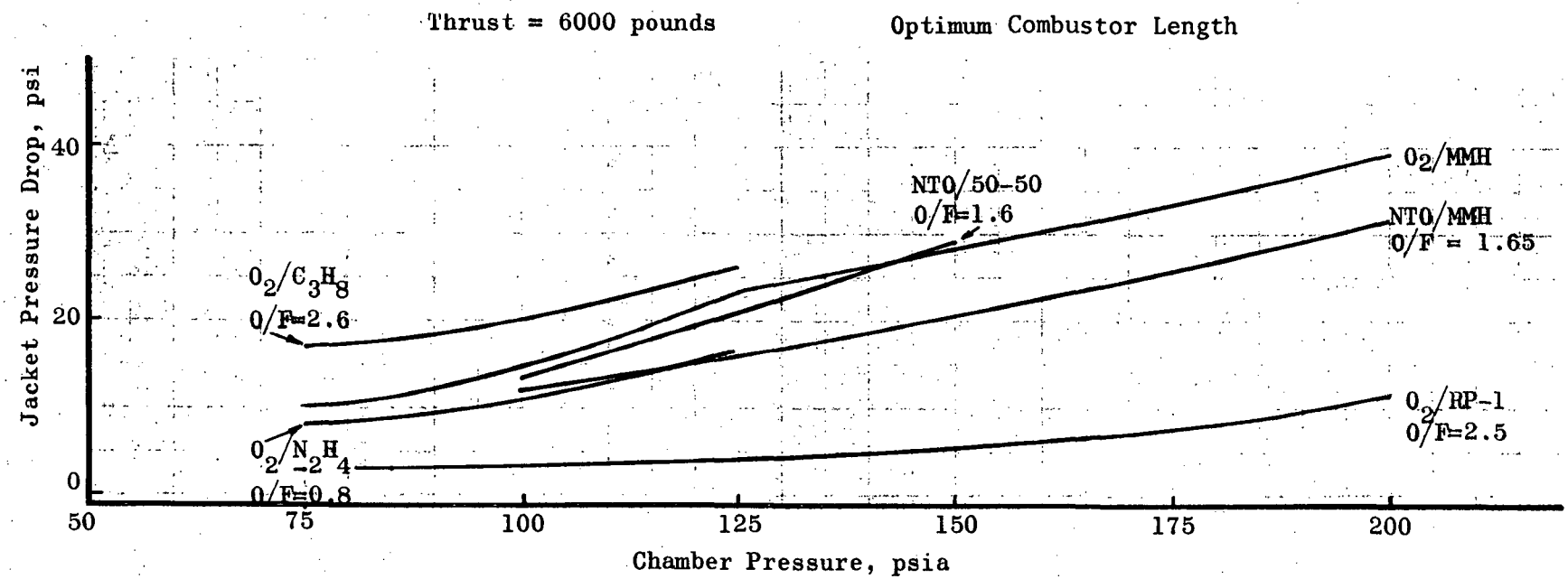


Figure 42. Regenerative Coolant Jacket Pressure Drop

Propellant	Injector	T/C Cooling	Curve
O_2, NTO	L-D, U-D	Regen, Film	— — — —
MMH, 50-50 RP-1, N_2H_4	L-D	Regen, Film	— — — —
MMH, 50-50	U-D	Regen	————
MMH, 50-50	U-D	Film	— - —

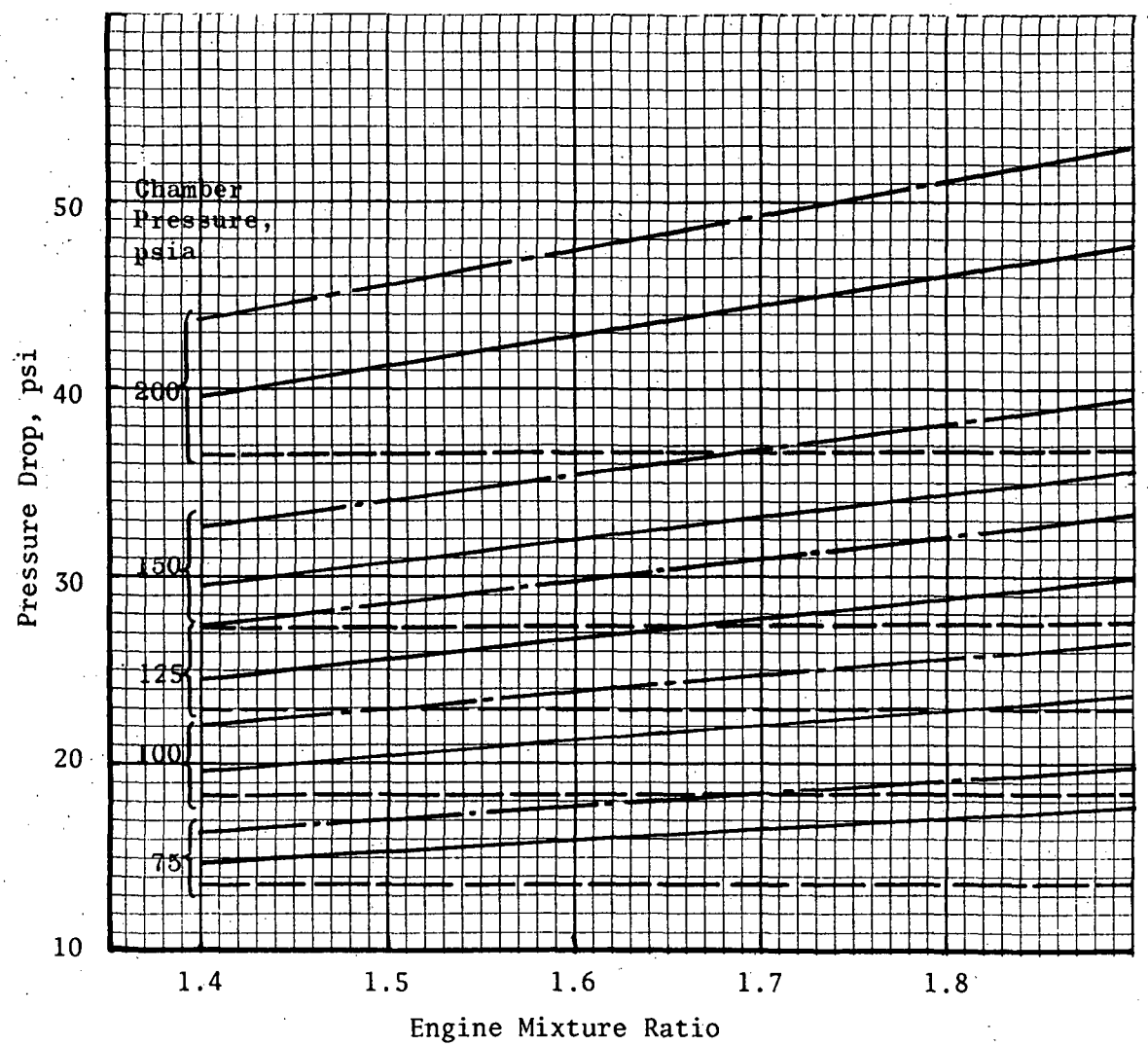


Figure 43. Injector Orifice Pressure Drop

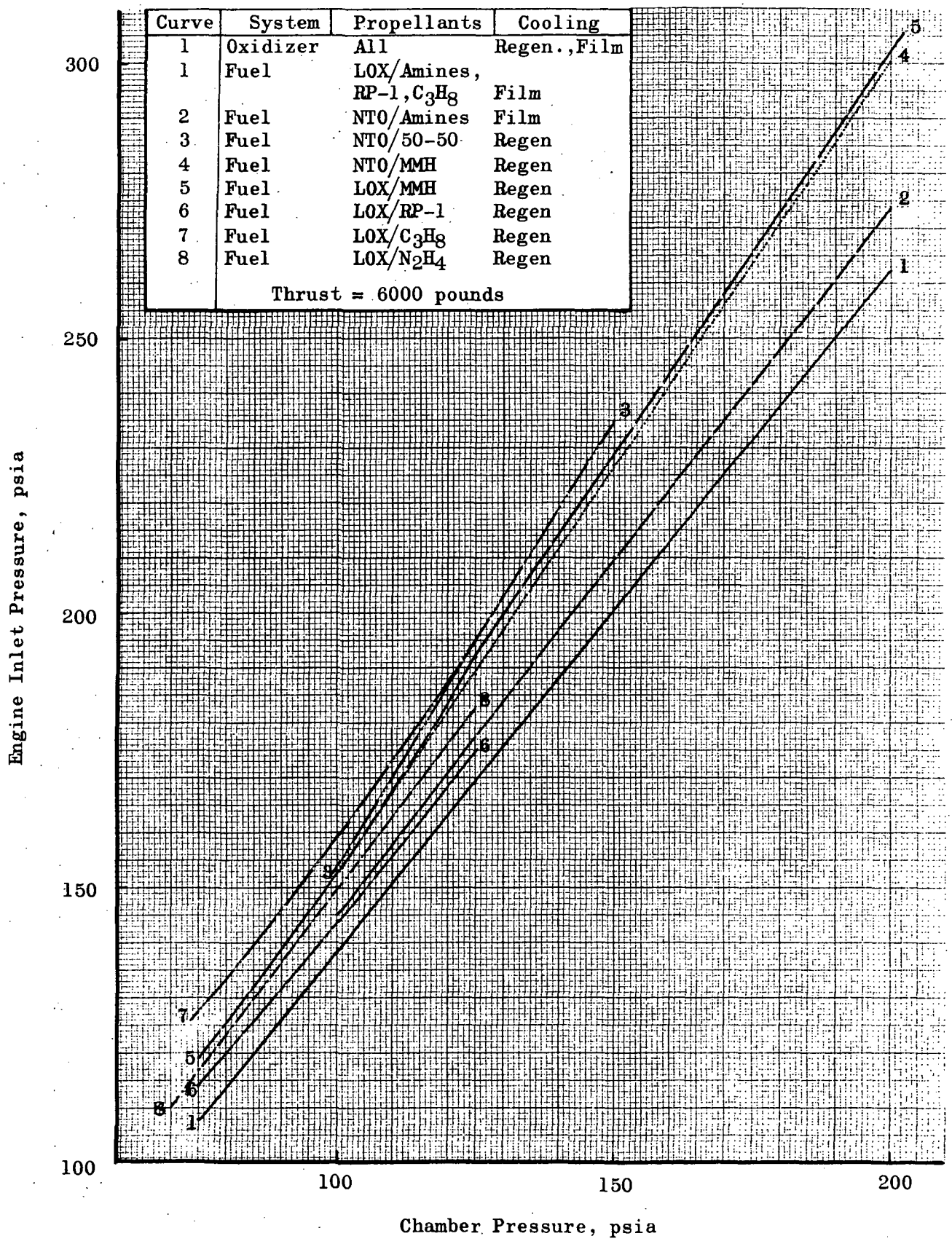


Figure 44. Engine Inlet Pressures

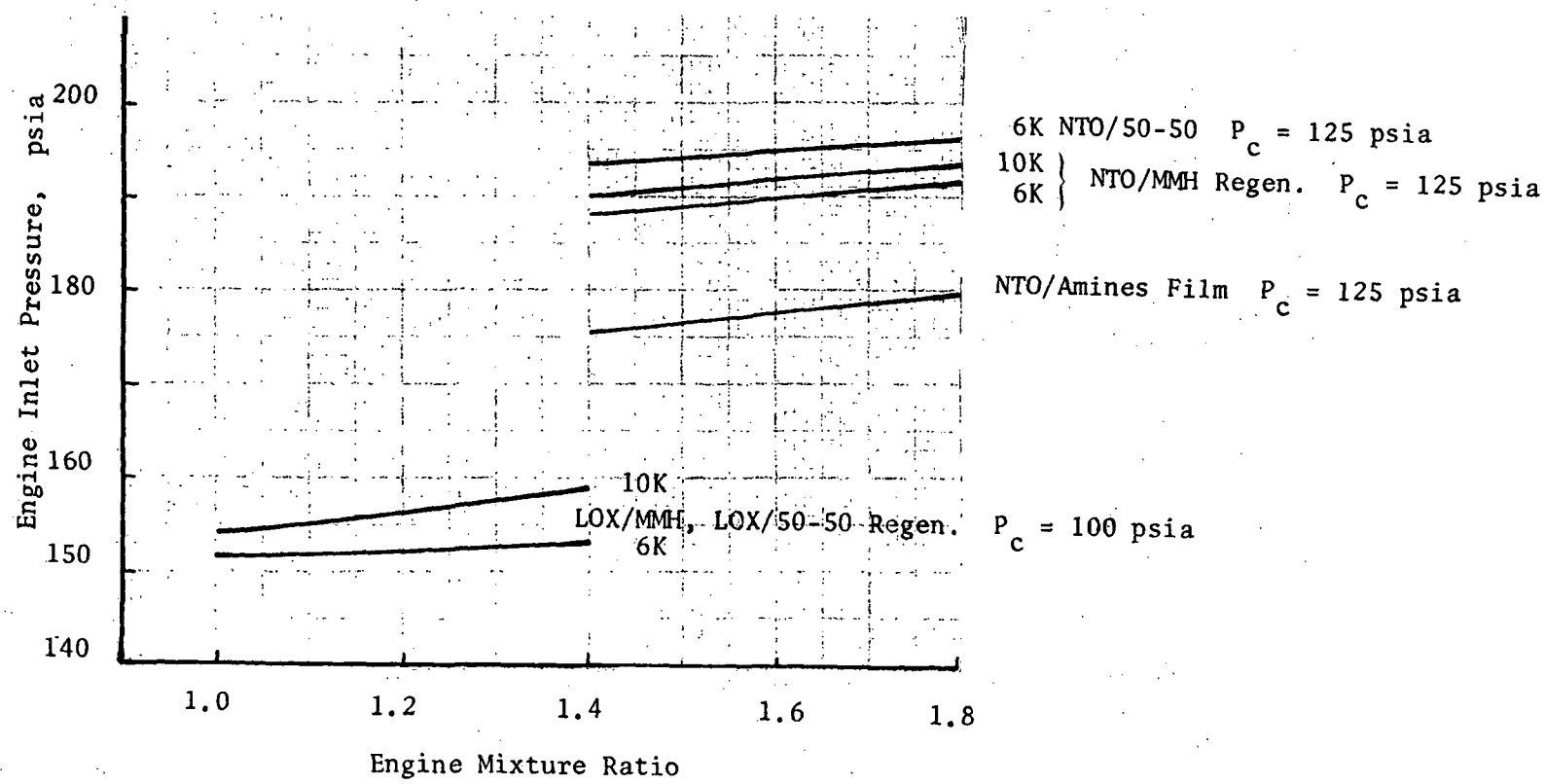


Figure 45. Mixture Ratio and Thrust Effect on Inlet Pressure

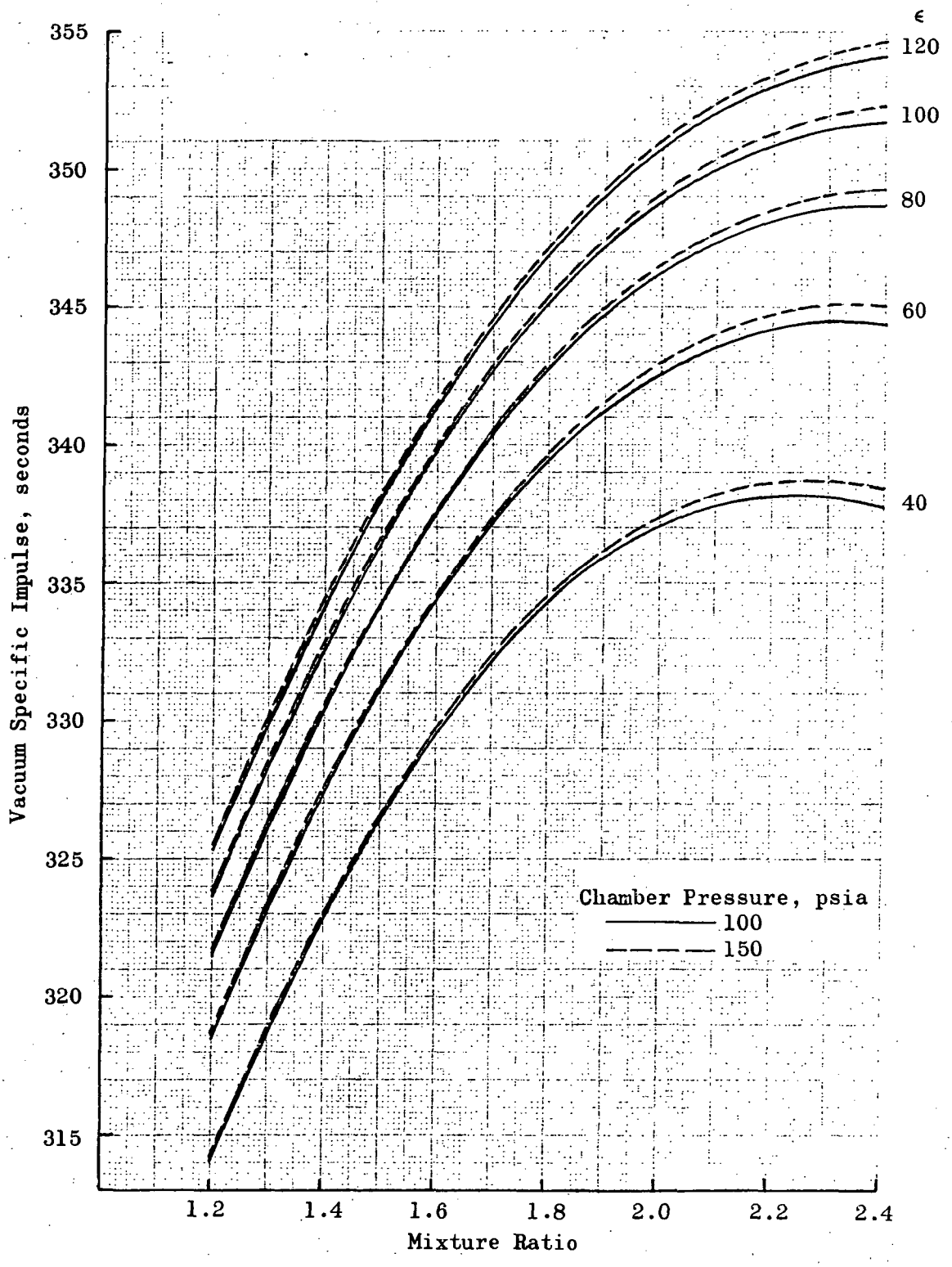


Figure 46. Theoretical One-Dimensional Equilibrium Specific Impulse, NTO/MMH

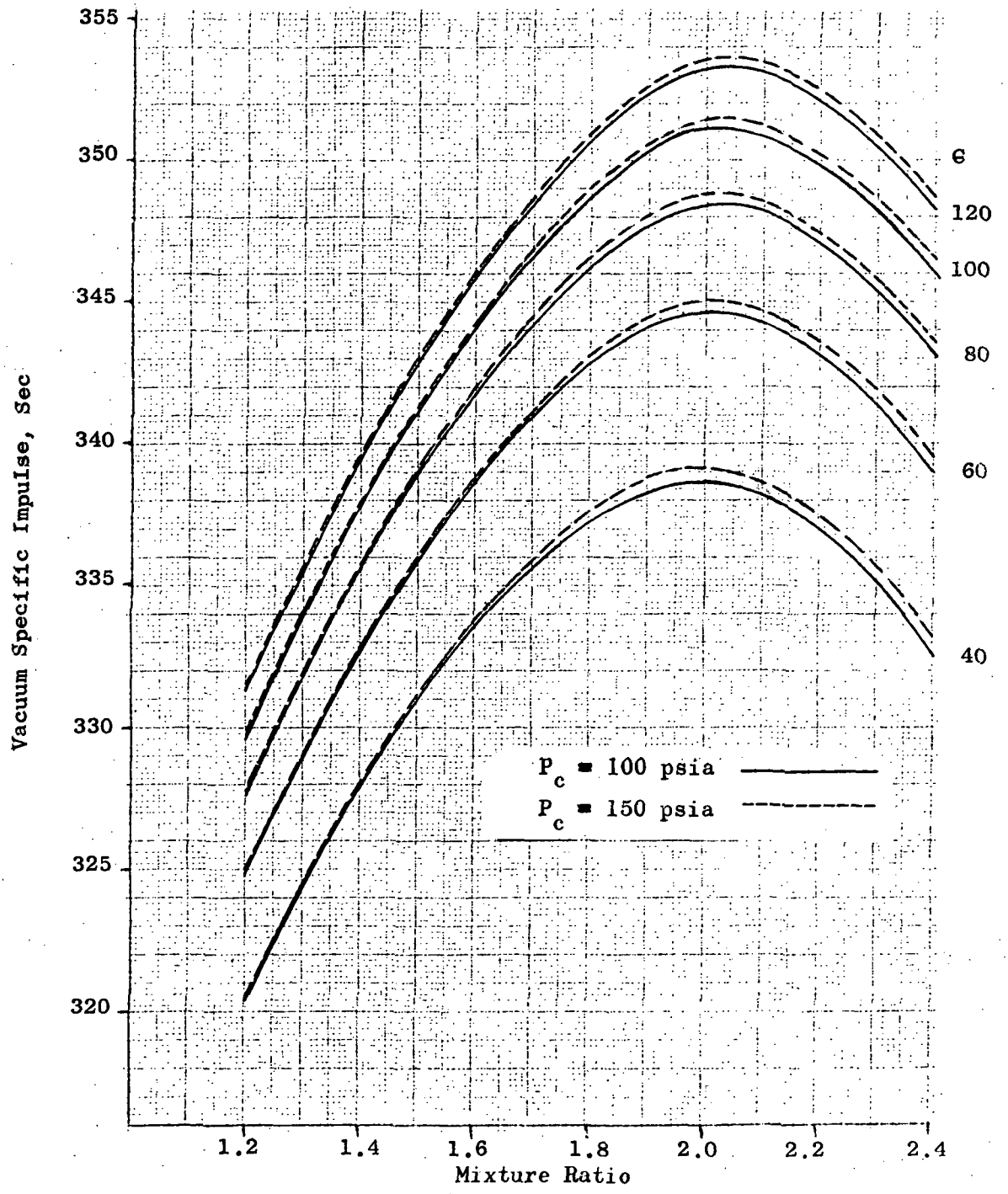


Figure 47. Theoretical One-Dimensional Equilibrium Specific Impulse, NTO/50-50.

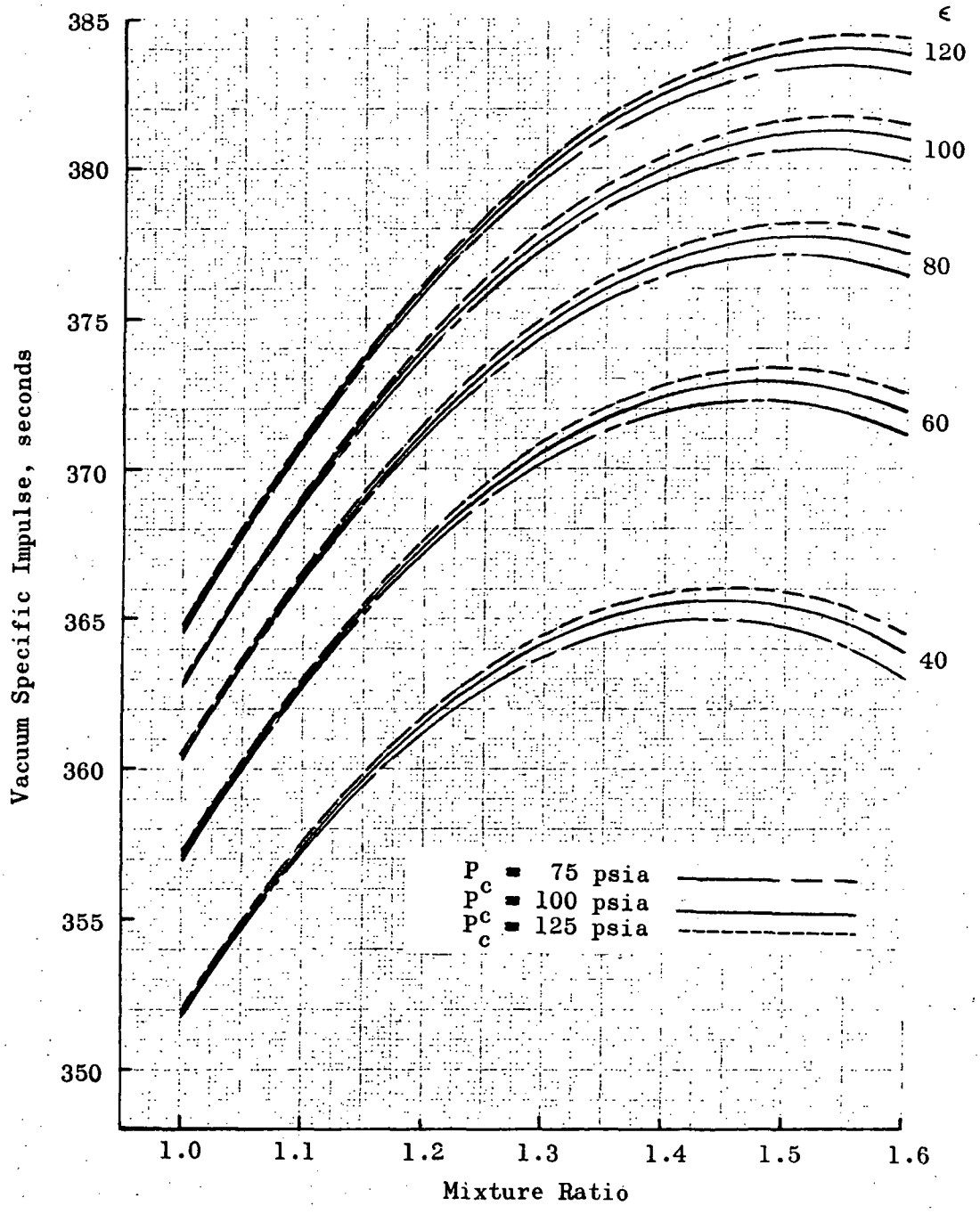


Figure 48. Theoretical One-Dimensional Equilibrium Specific Impulse, O_2/MMH

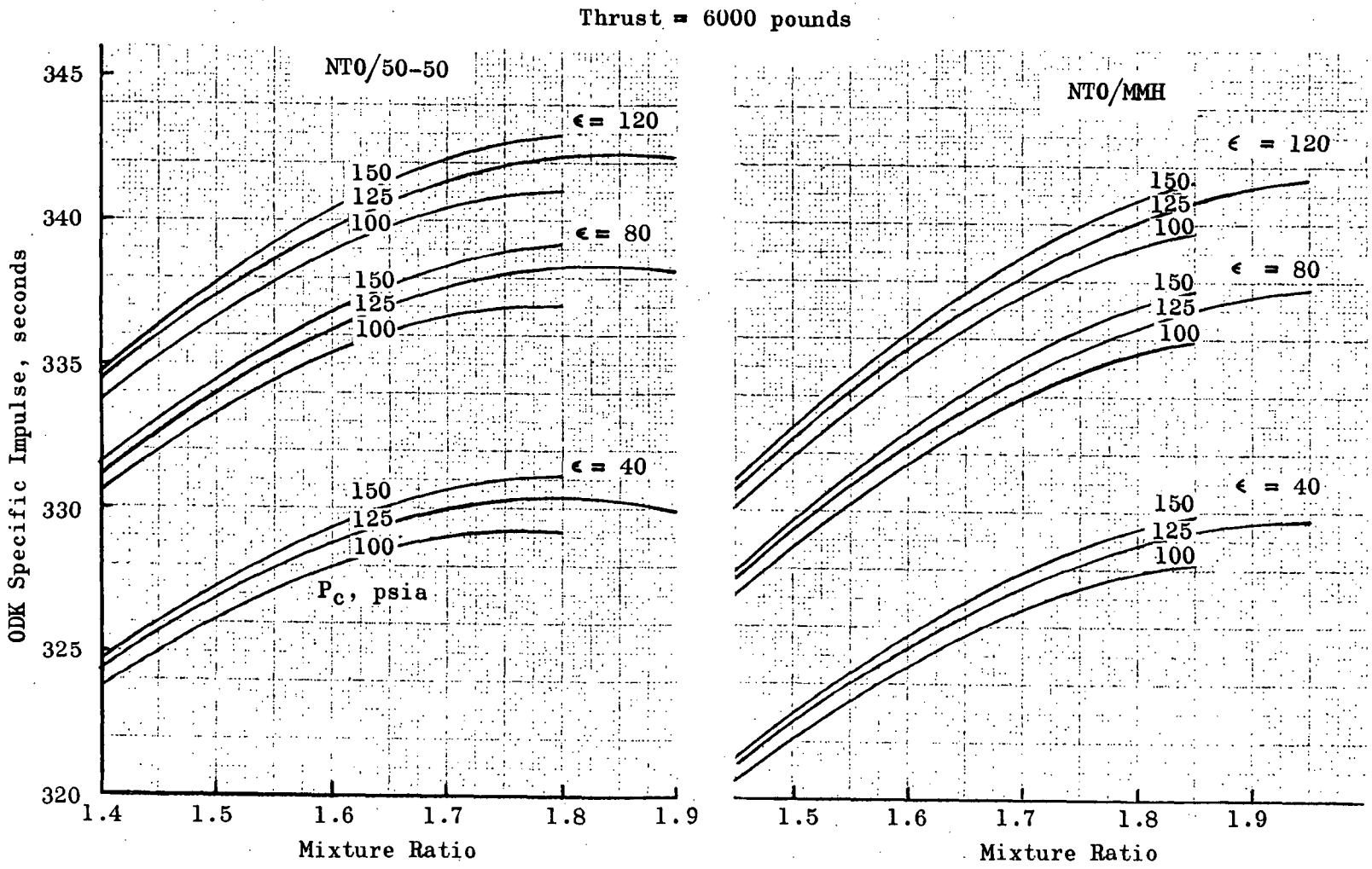


Figure 49. Theoretical ODK Performance for NTO/50-50 and NTO/MMH

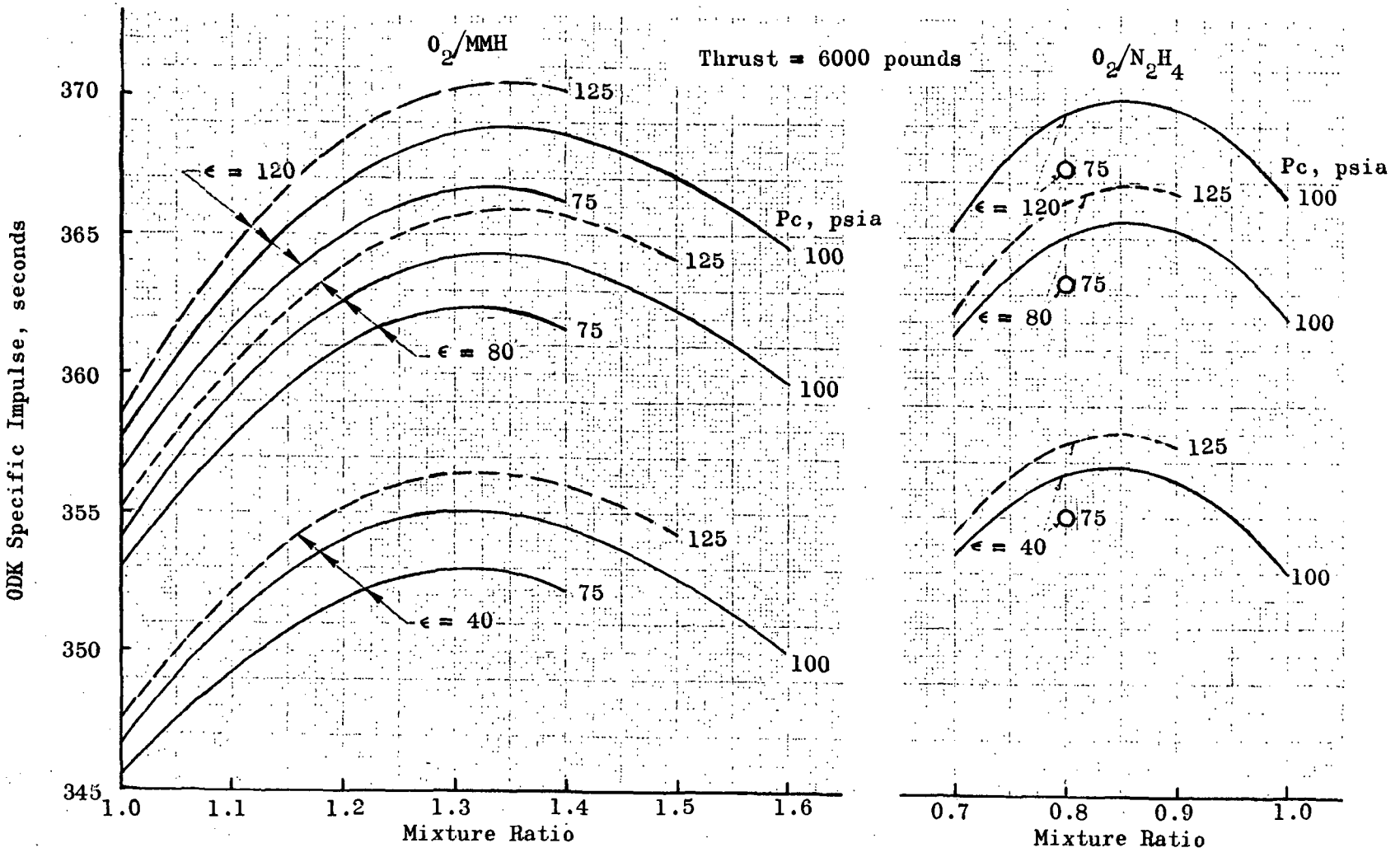


Figure 50. Theoretical ODK Performance for O_2/MMH and O_2/N_2H_4

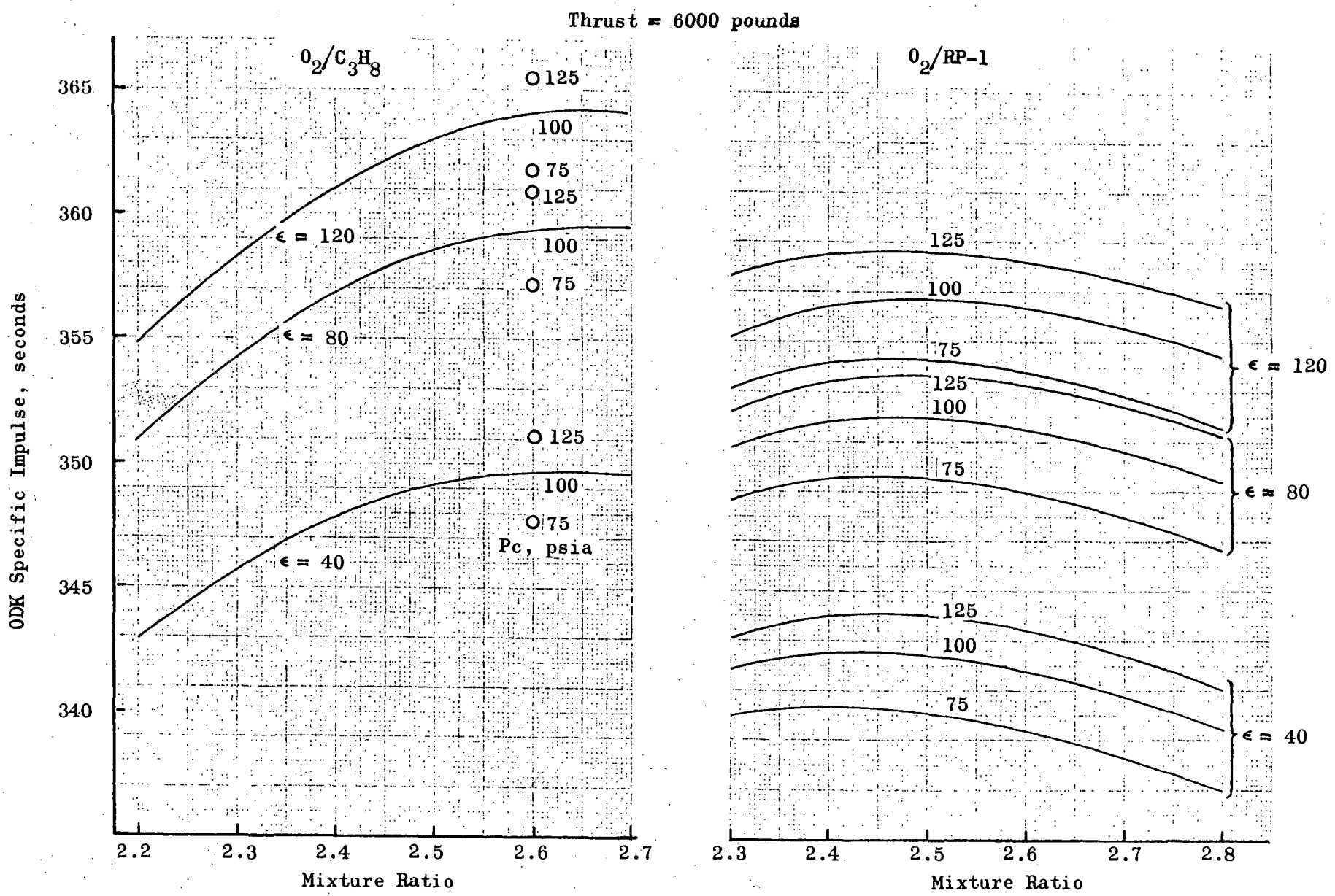


Figure 51. Theoretical ODK Performance for O_2/C_3H_8 and O_2/RP-1

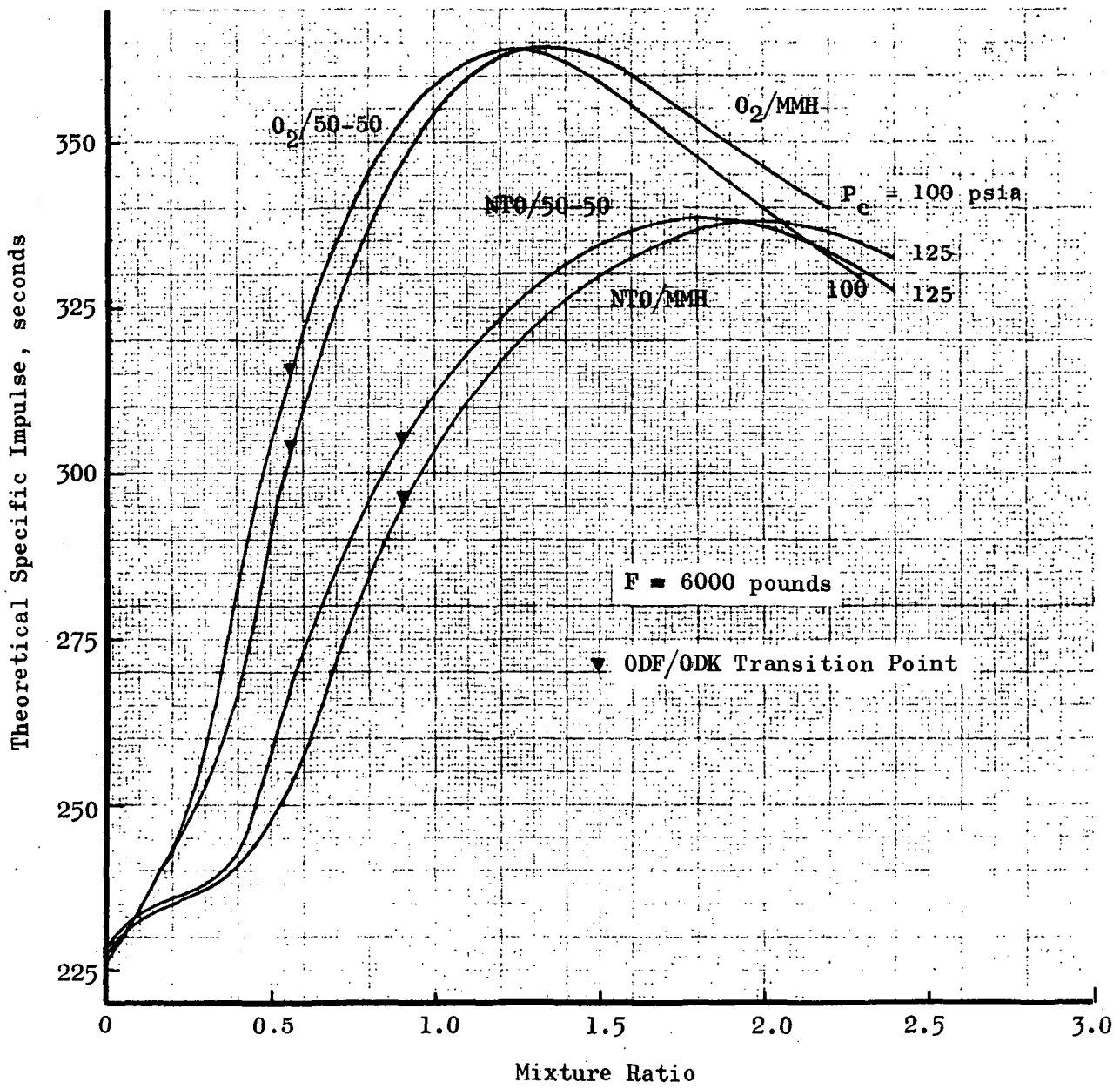
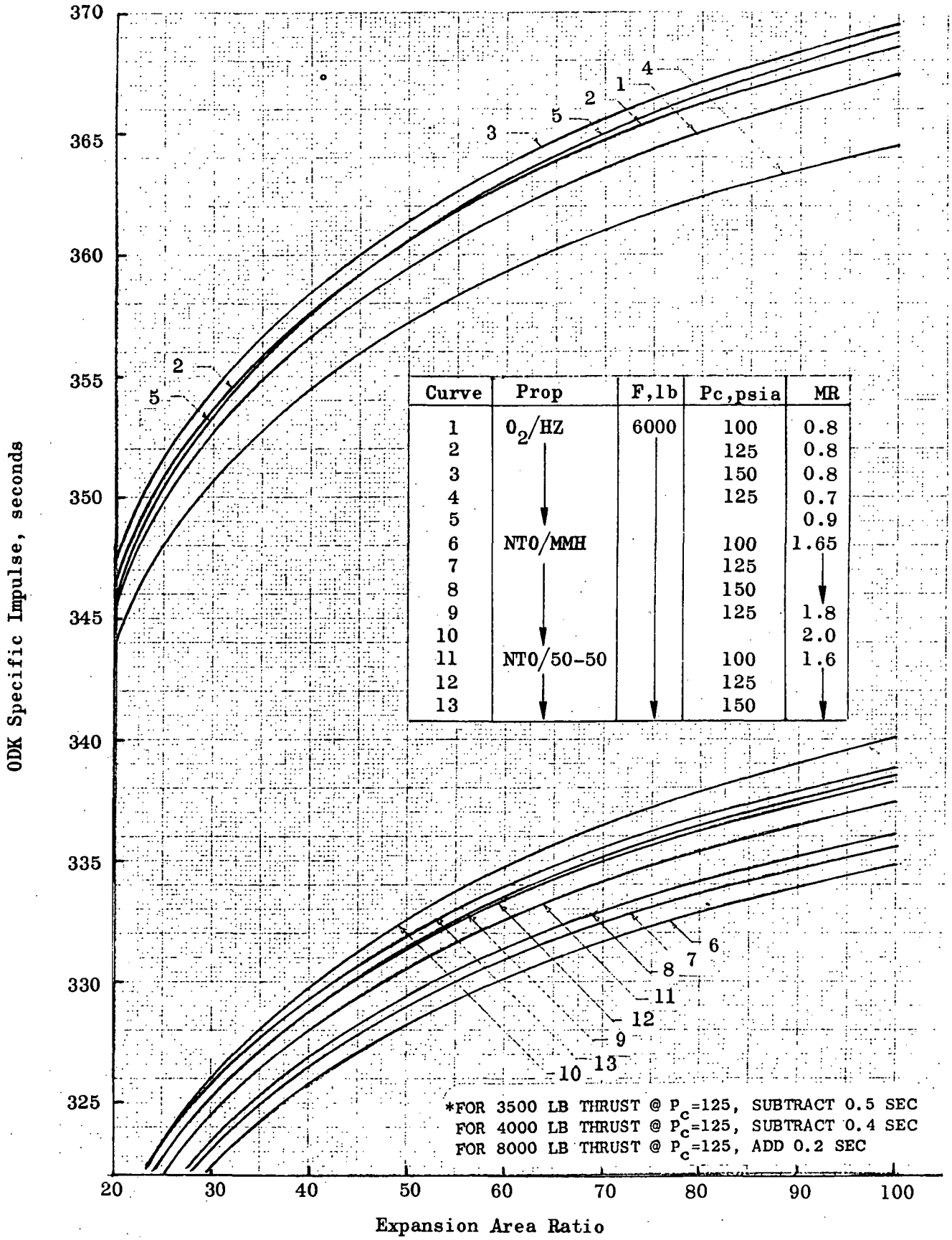
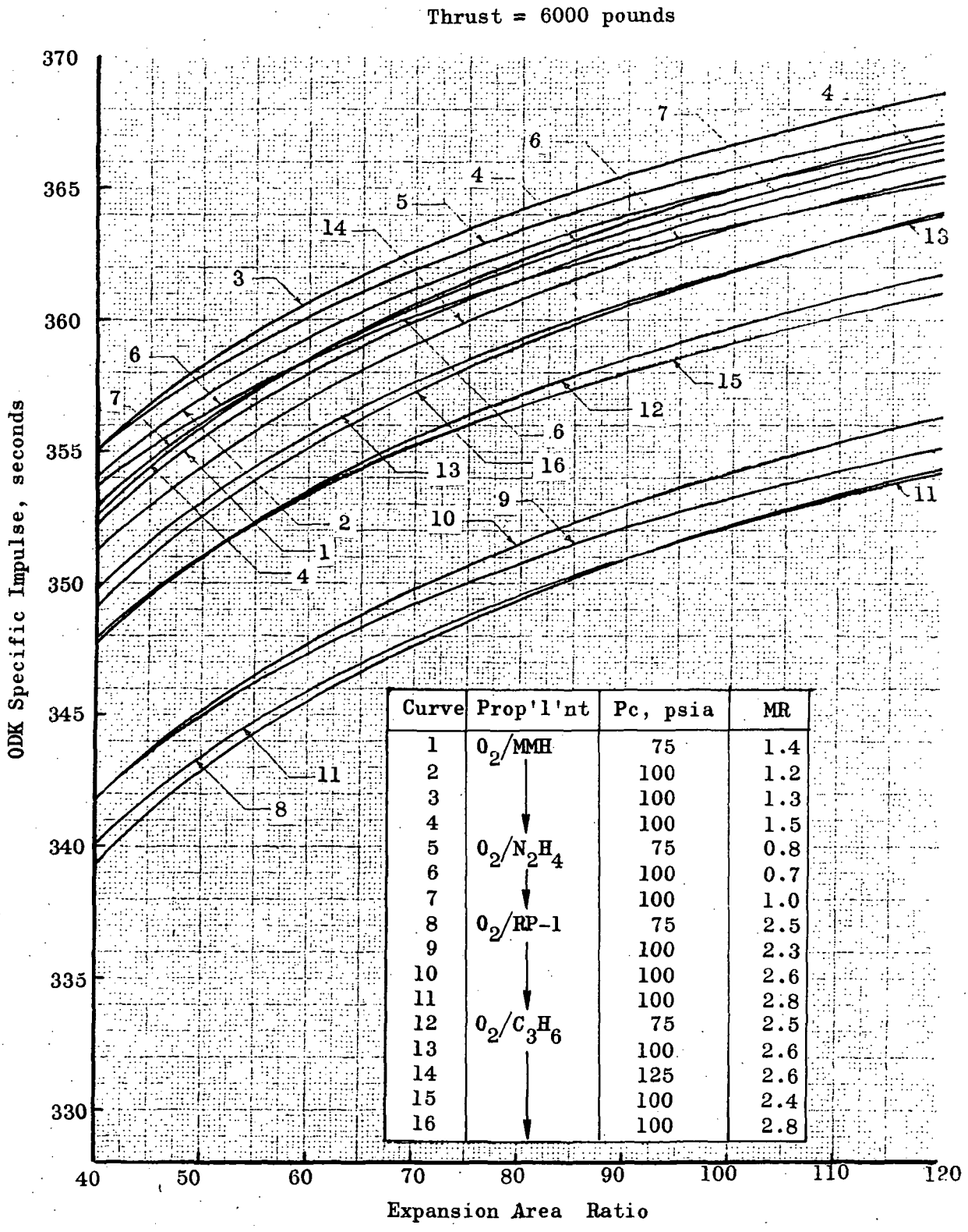


Figure 52. Performance of Carbon Containing Amines at Low o/f



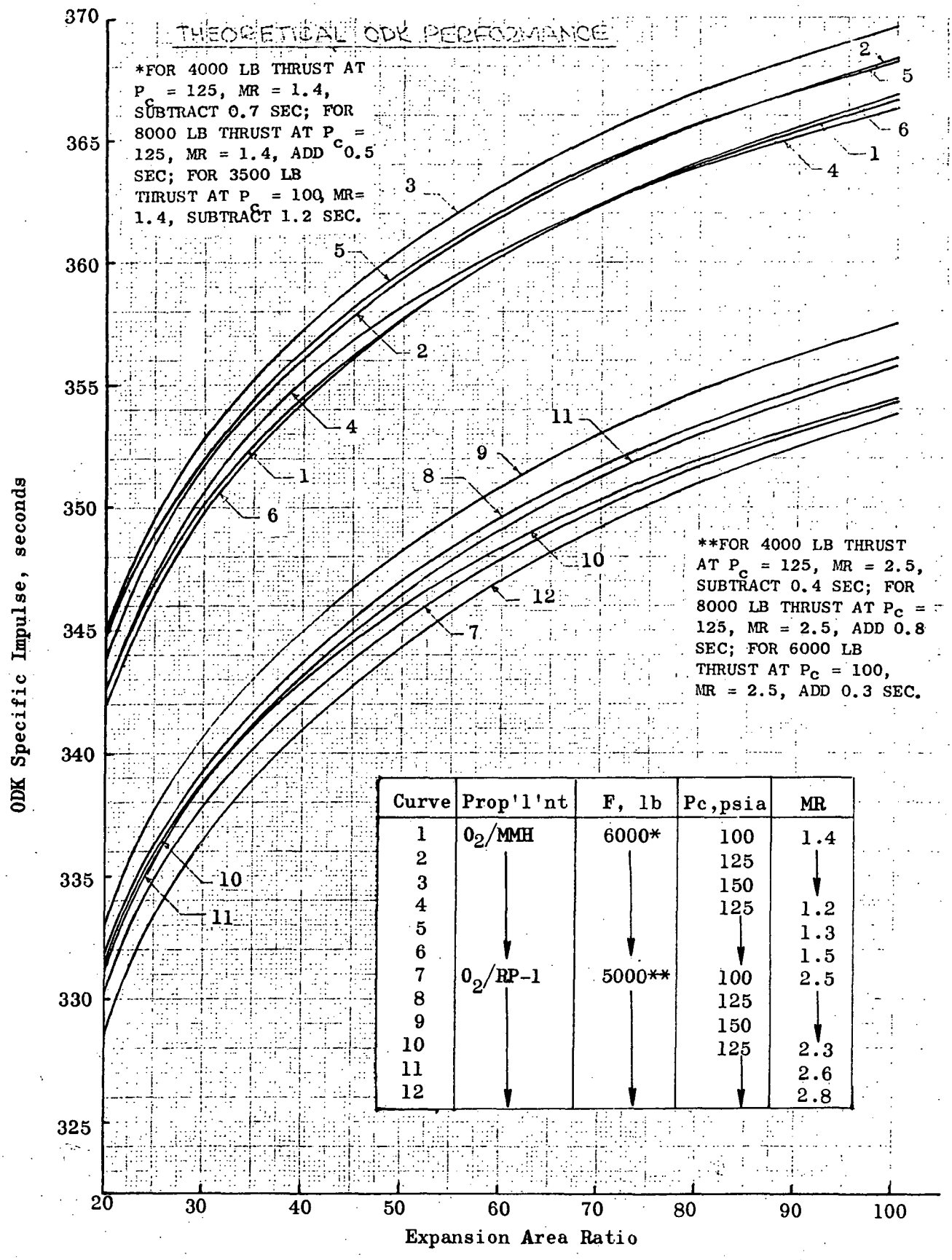
Curve	Prop	F,lb	Pc,psia	MR
1	O_2/HZ	6000	100	0.8
2			125	0.8
3			150	0.8
4			125	0.7
5				0.9
6	NTO/MMH		100	1.65
7			125	
8			150	
9			125	1.8
10				2.0
11	NTO/50-50		100	1.6
12			125	
13			150	

Figure 53. ODK Specific Impulse vs Area Ratio



Curve	Prop'l'nt	Pc, psia	MR
1	O_2/MMH	75	1.4
2		100	1.2
3		100	1.3
4		100	1.5
5	O_2/N_2H_4	75	0.8
6		100	0.7
7		100	1.0
8	O_2/RP-1	75	2.5
9		100	2.3
10		100	2.6
11		100	2.8
12	O_2/C_3H_6	75	2.5
13		100	2.6
14		125	2.6
15		100	2.4
16		100	2.8

Figure 54. Theoretical ODK Performance



Curve	Prop'l'nt	F, lb	Pc,psia	MR
1	O_2/MMH	6000*	100	1.4
2			125	
3			150	
4			125	1.2
5				1.3
6				1.5
7	O_2/RP-1	5000**	100	2.5
8			125	
9			150	
10			125	2.3
11				2.6
12				2.8

Figure 55. ODK Specific Impulse vs Area Ratio

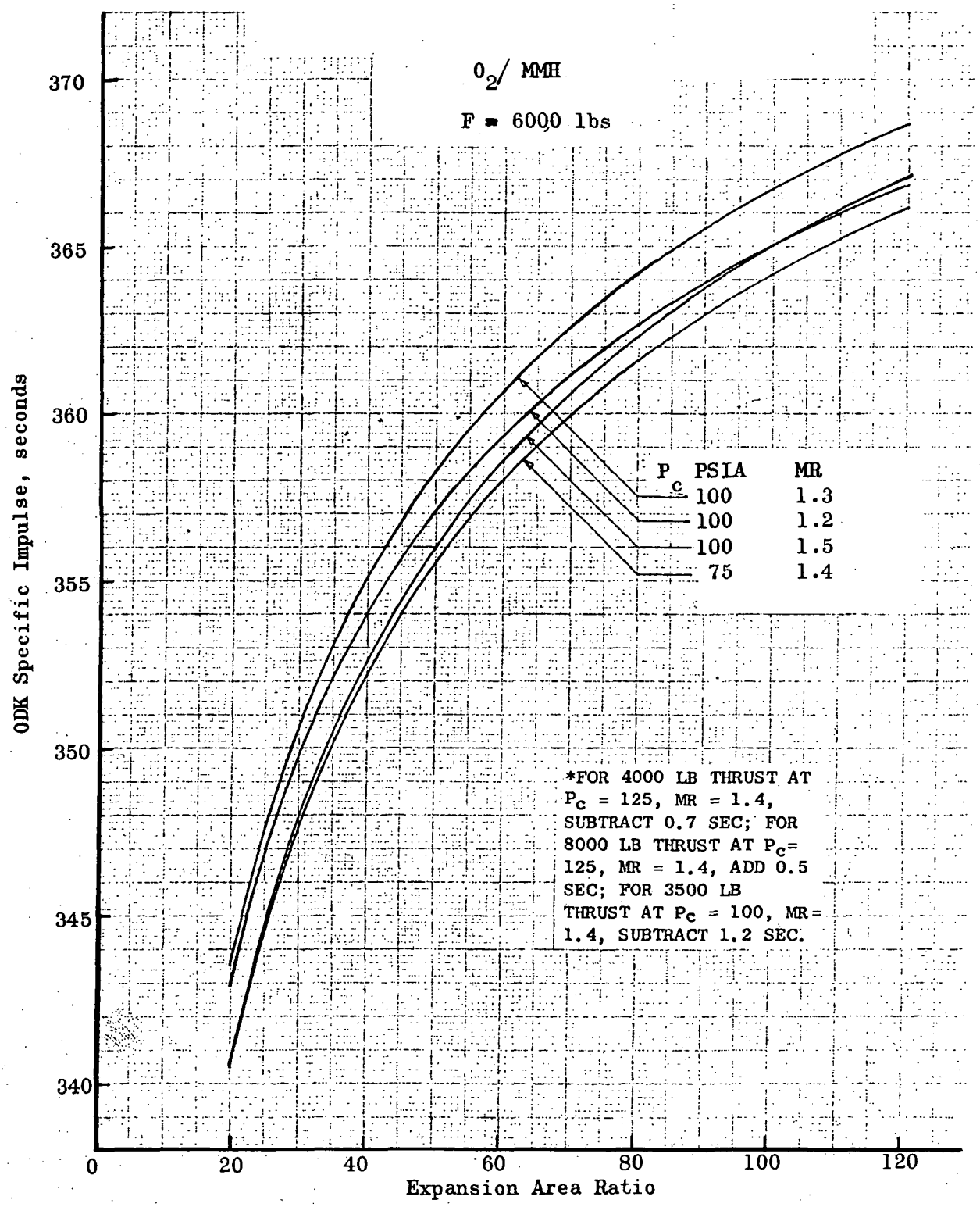


Figure 56. ODK Specific Impulse vs Area Ratio

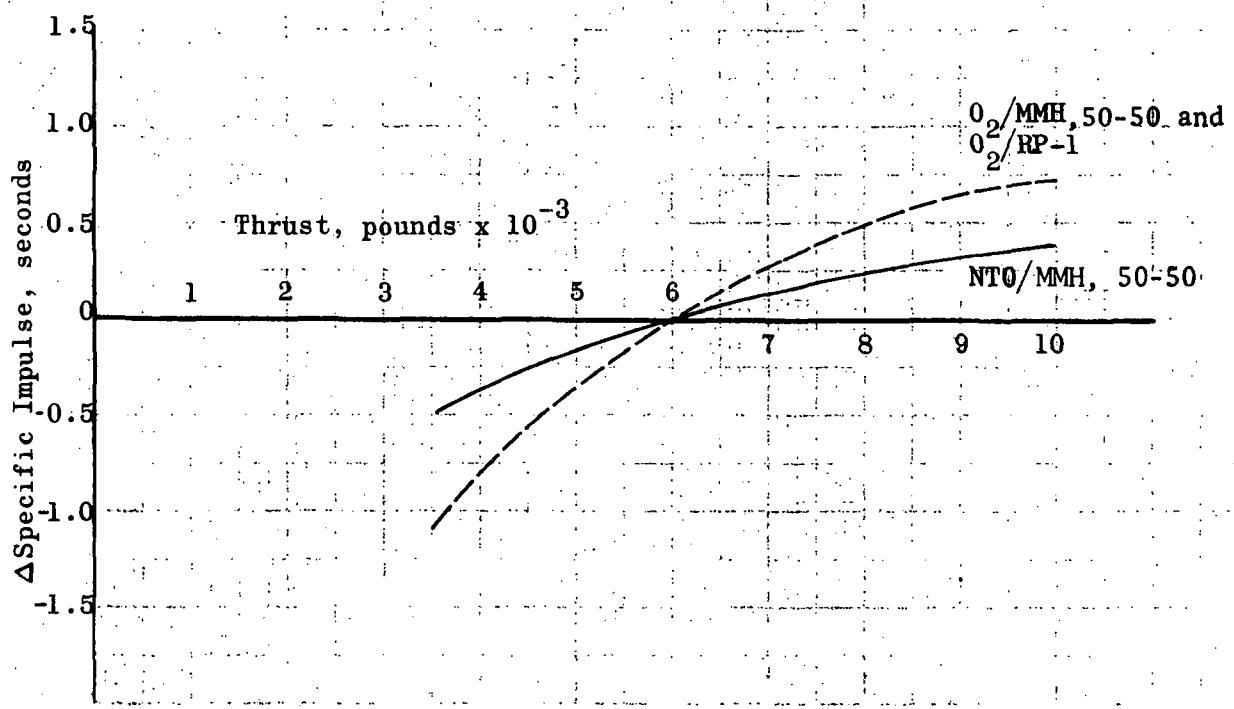


Figure 57. Effect of Thrust Level on Theoretical ODK Performance

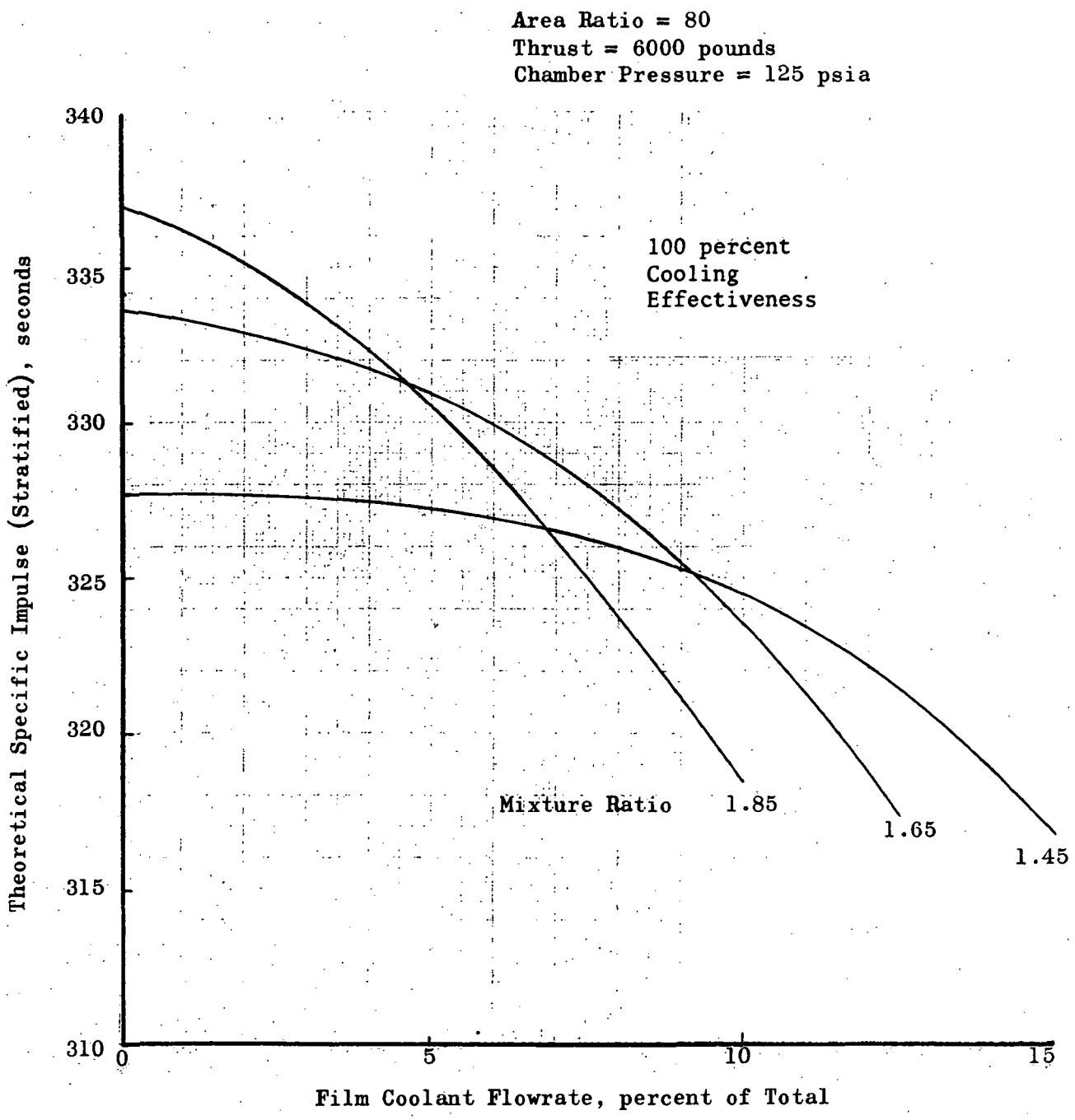


Figure 58. Stratified Performance for NTO/MMH

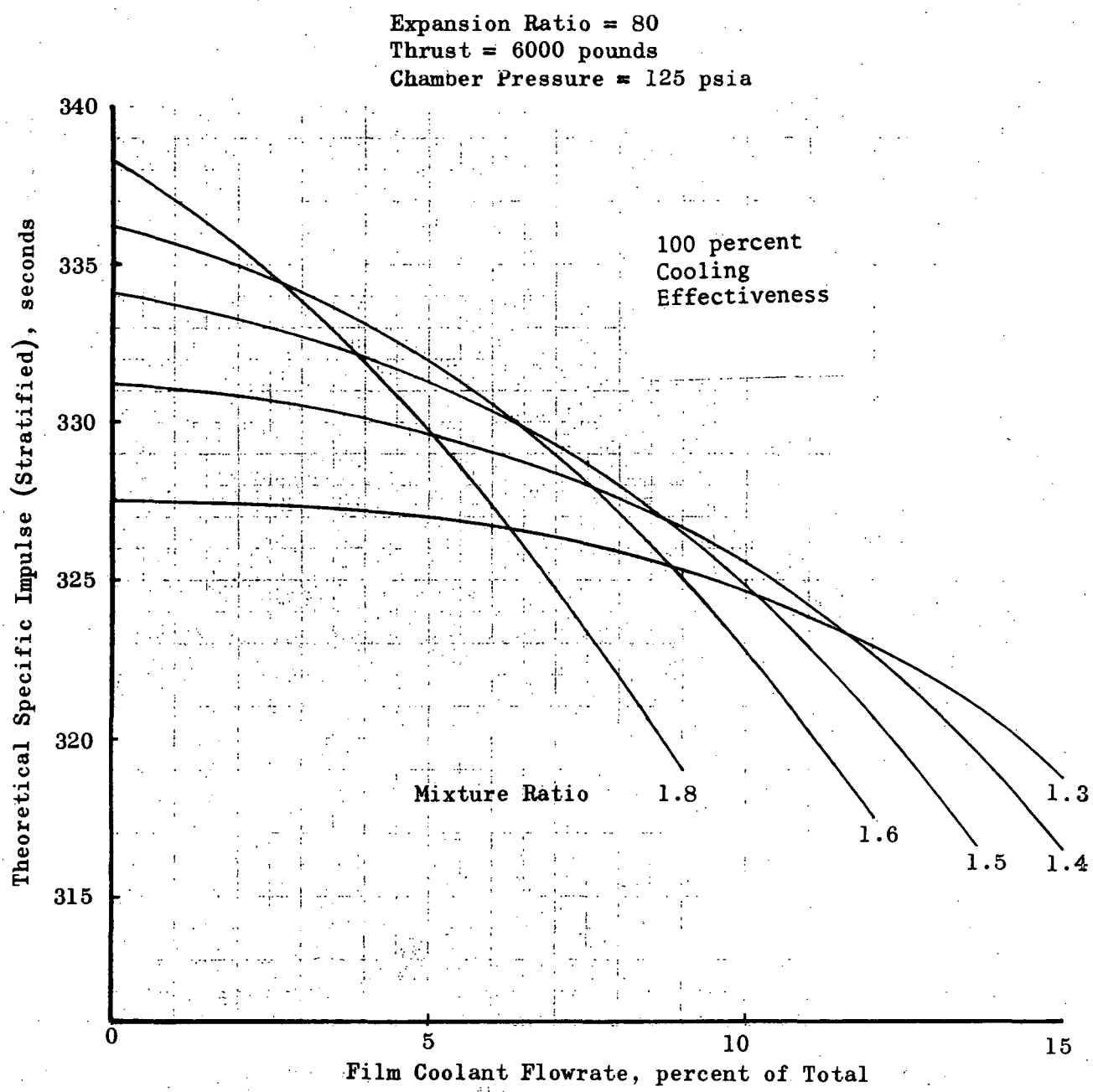


Figure 59. Stratified Performance for NTO/50-50

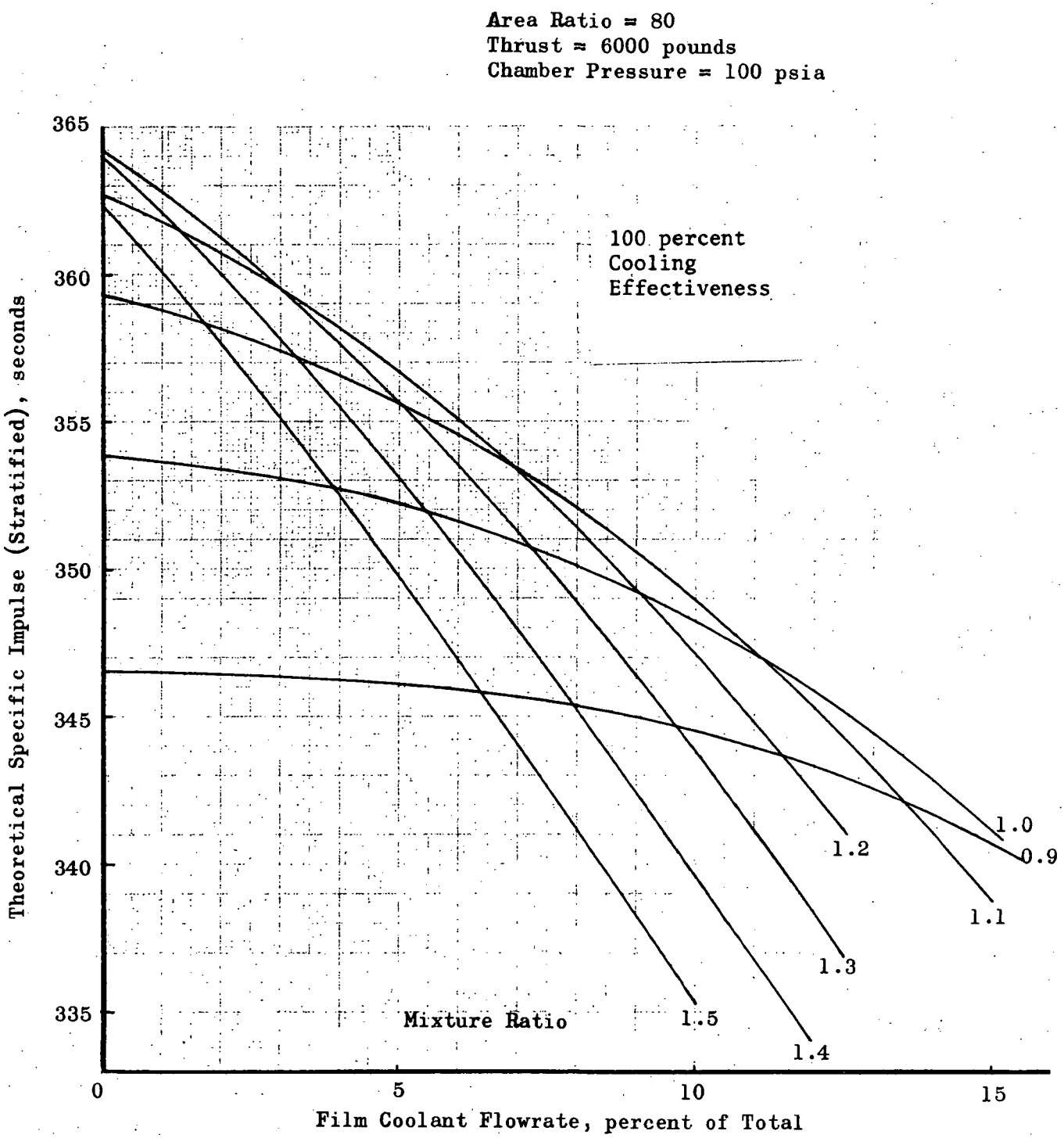


Figure 60. Stratified Performance for O_2/MMH

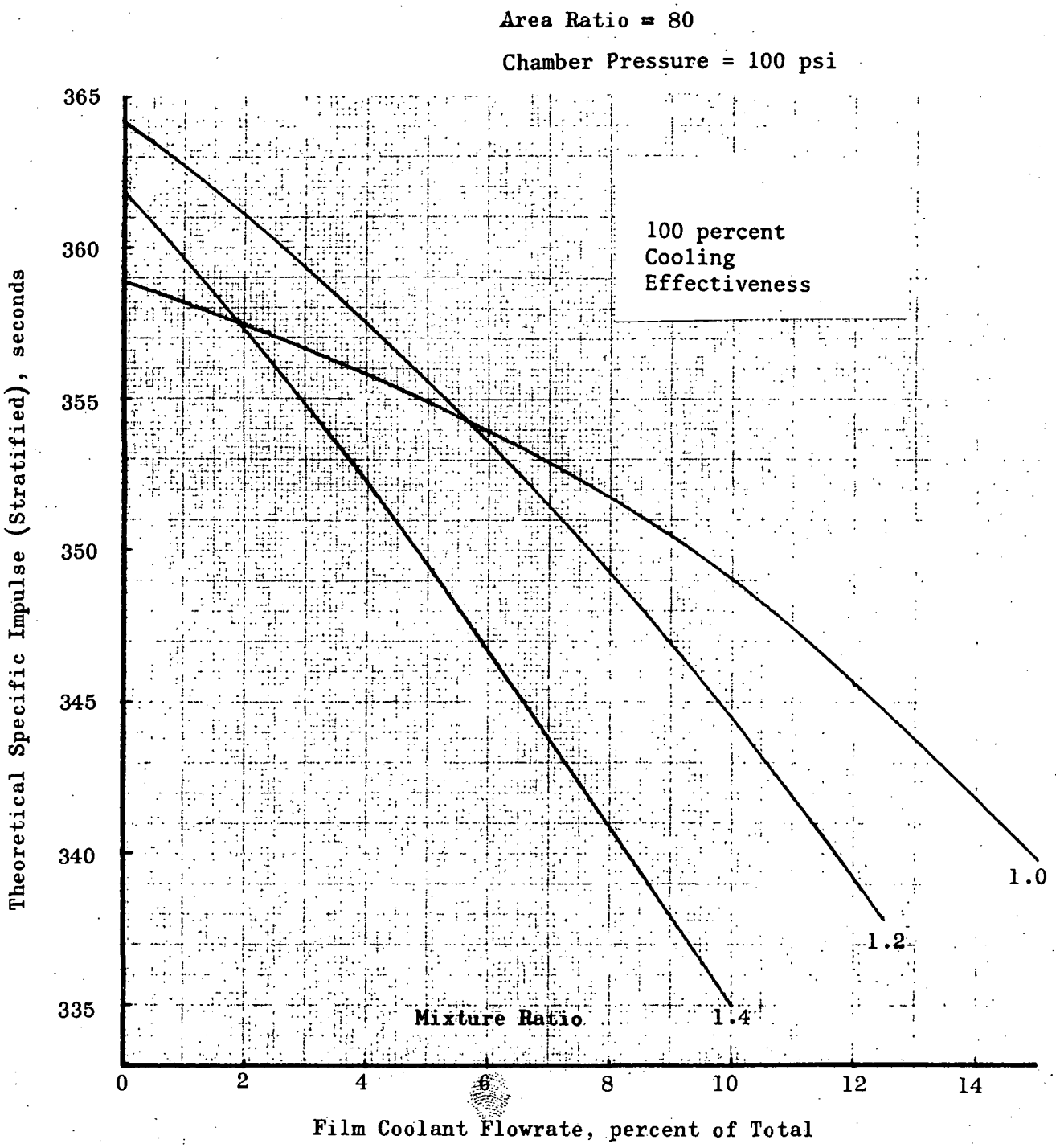
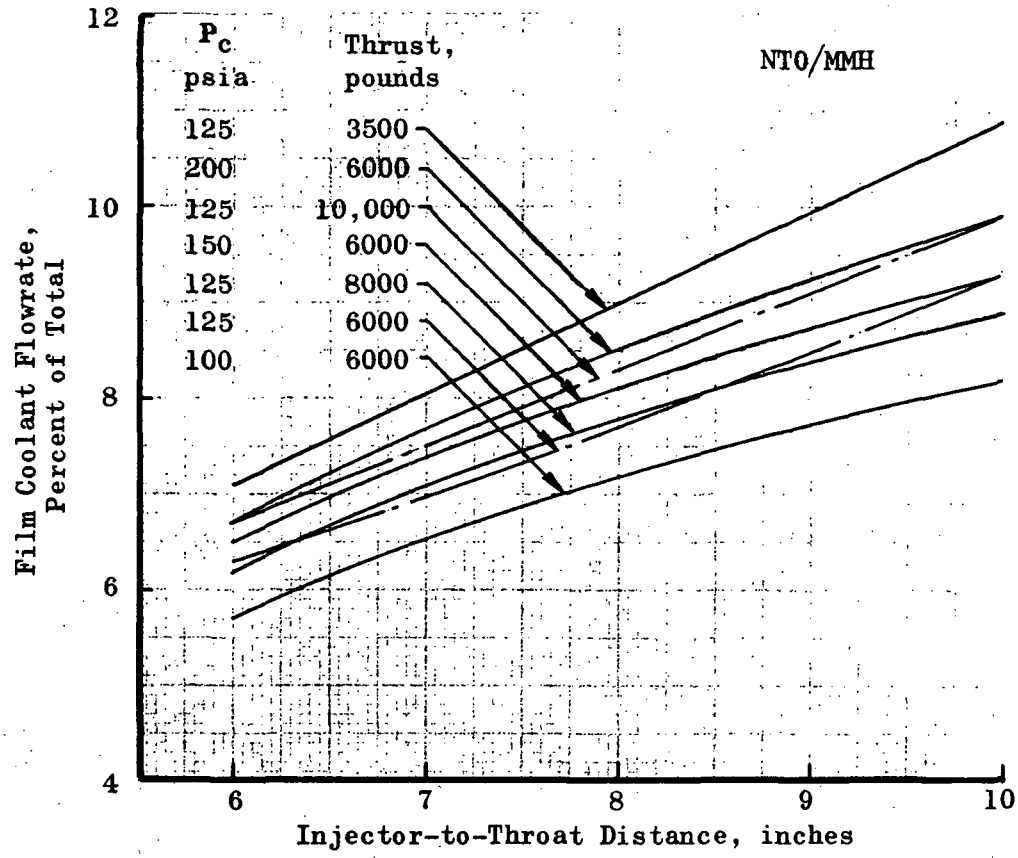


Figure 61. Stratified Performance for O_2/50-50



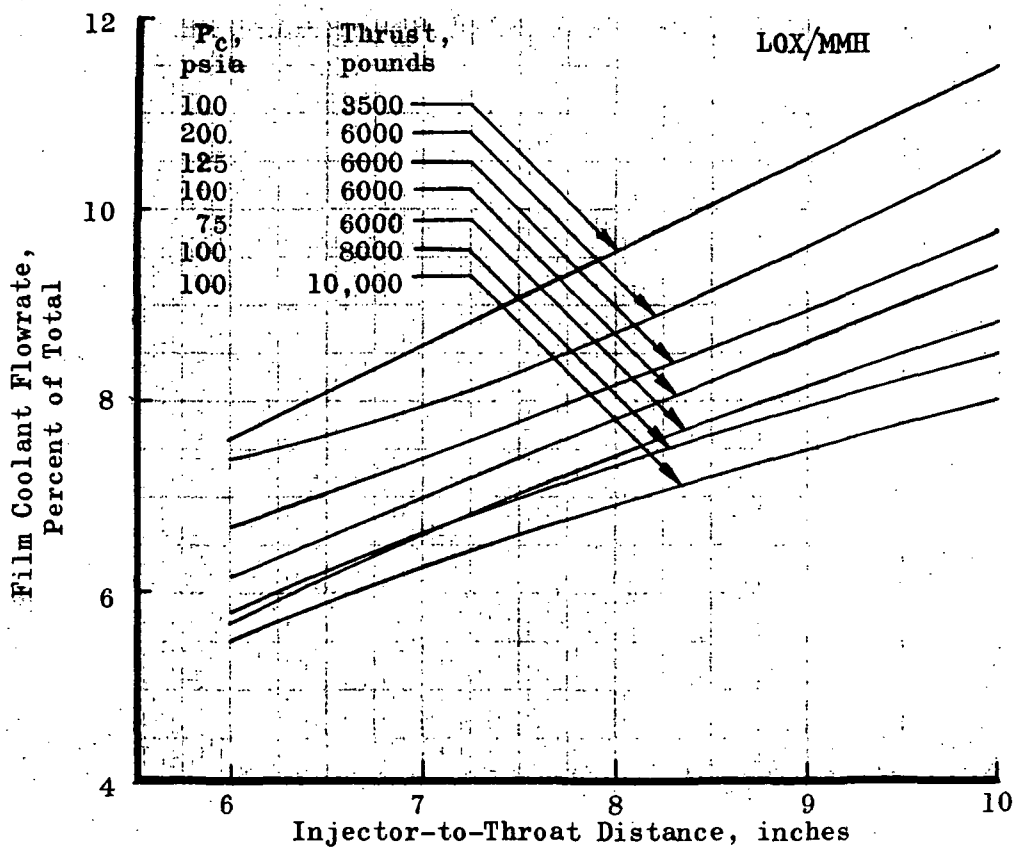


Figure 62. Film Coolant Flow in Dump/Film Cooled Thrust Chambers

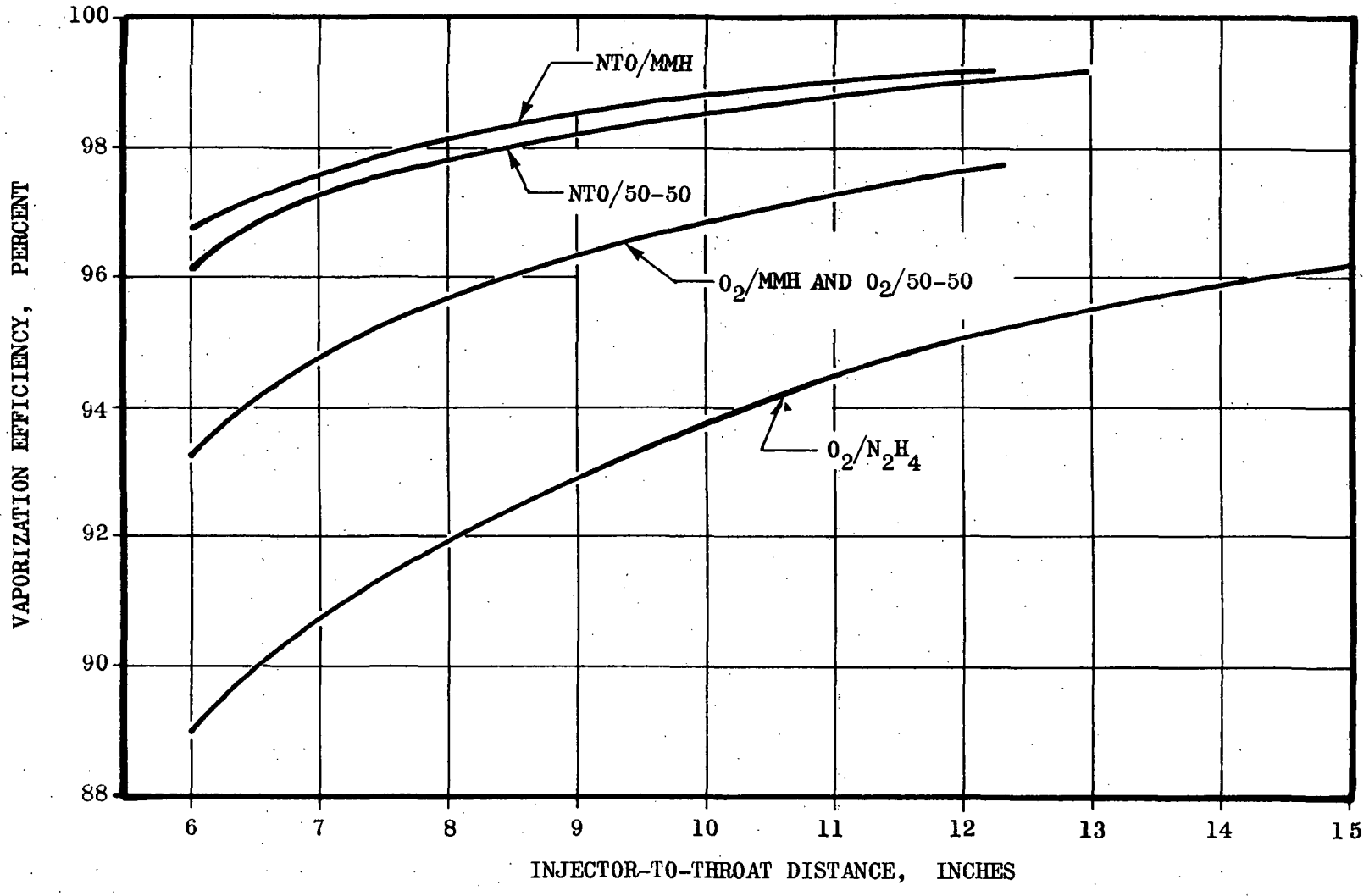


Figure 63. Injector Vaporization Efficiencies

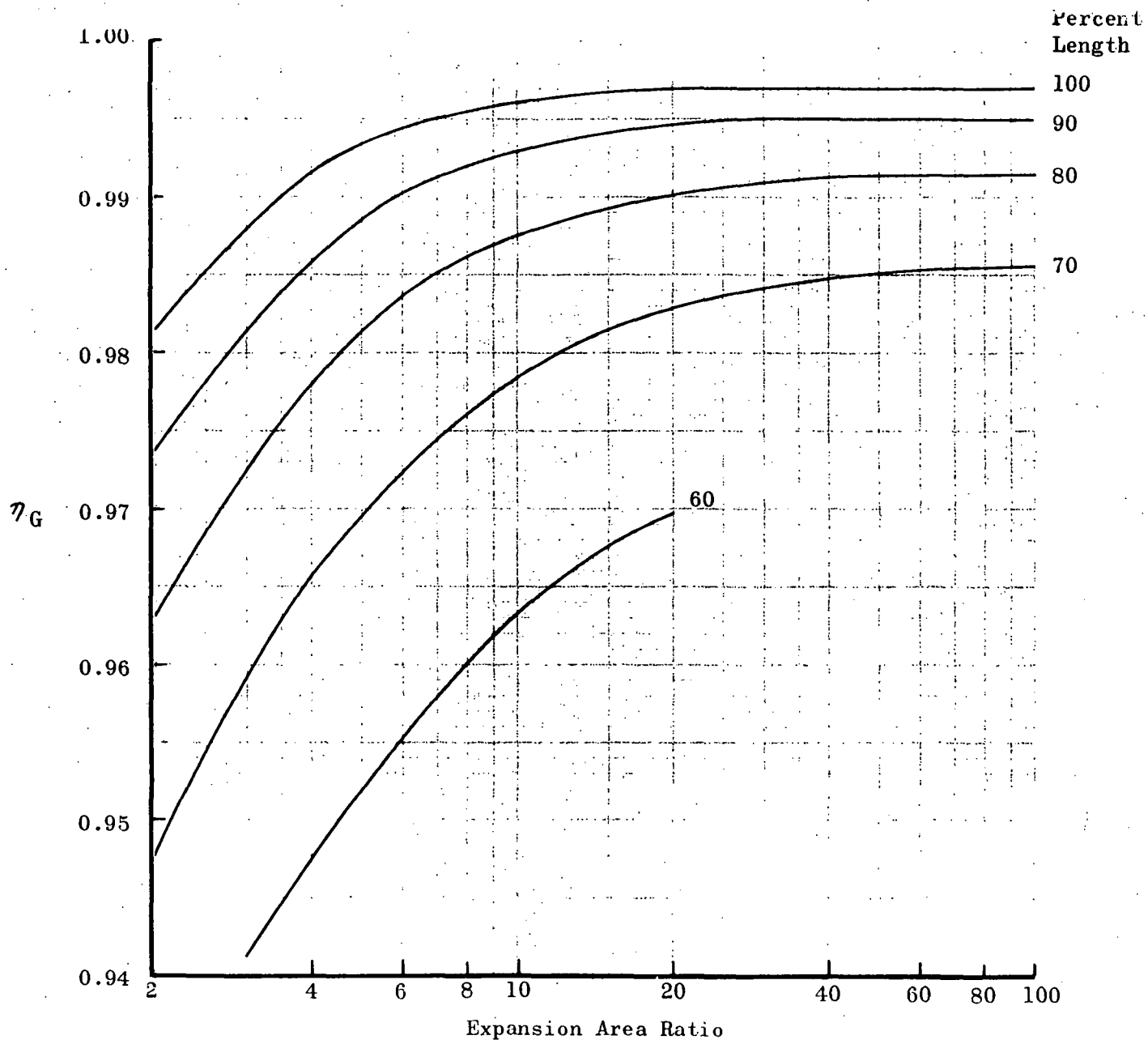


Figure 64. Parabolic Bell Nozzle Geometric Losses

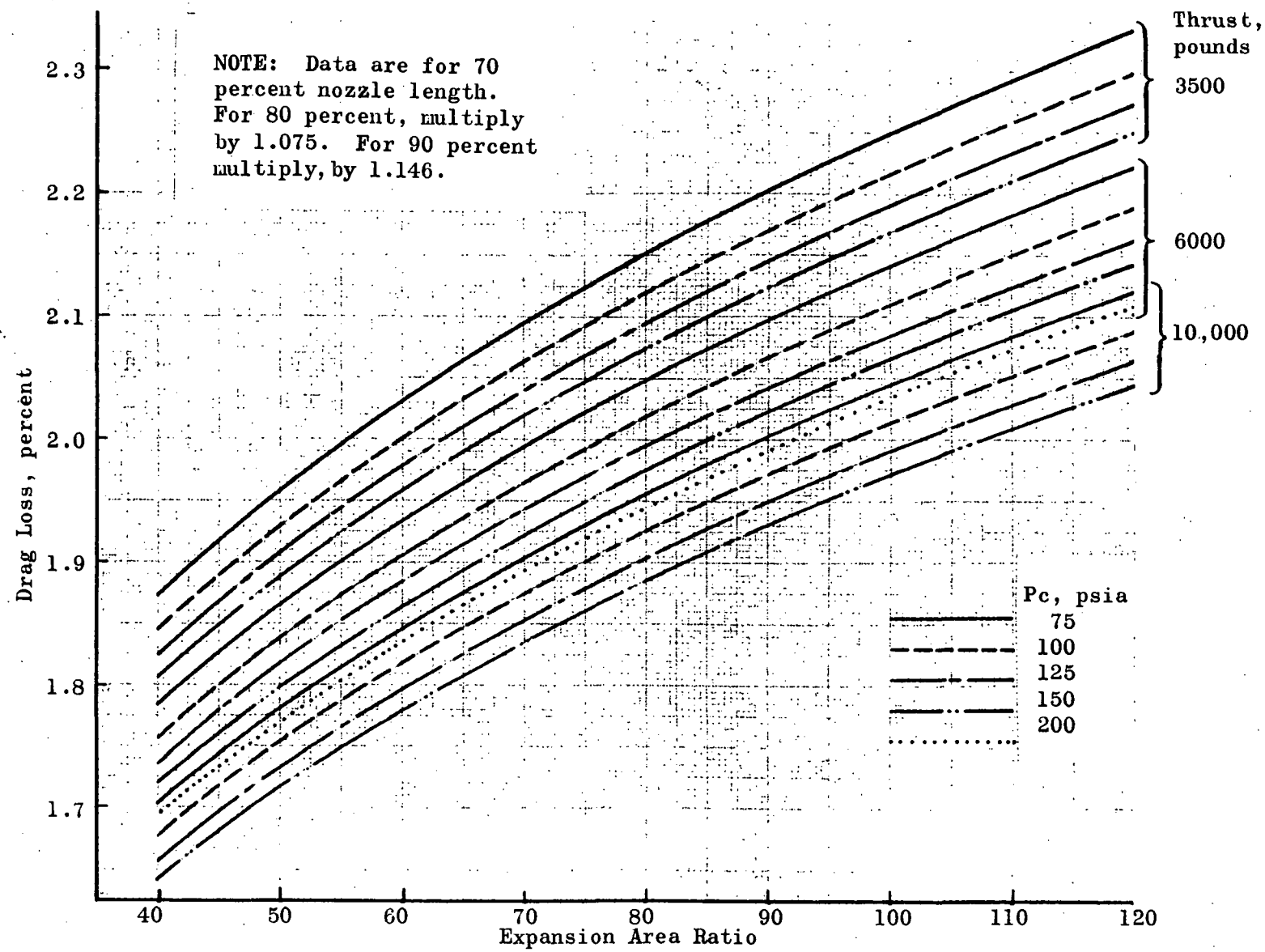


Figure 65. OME Drag Losses

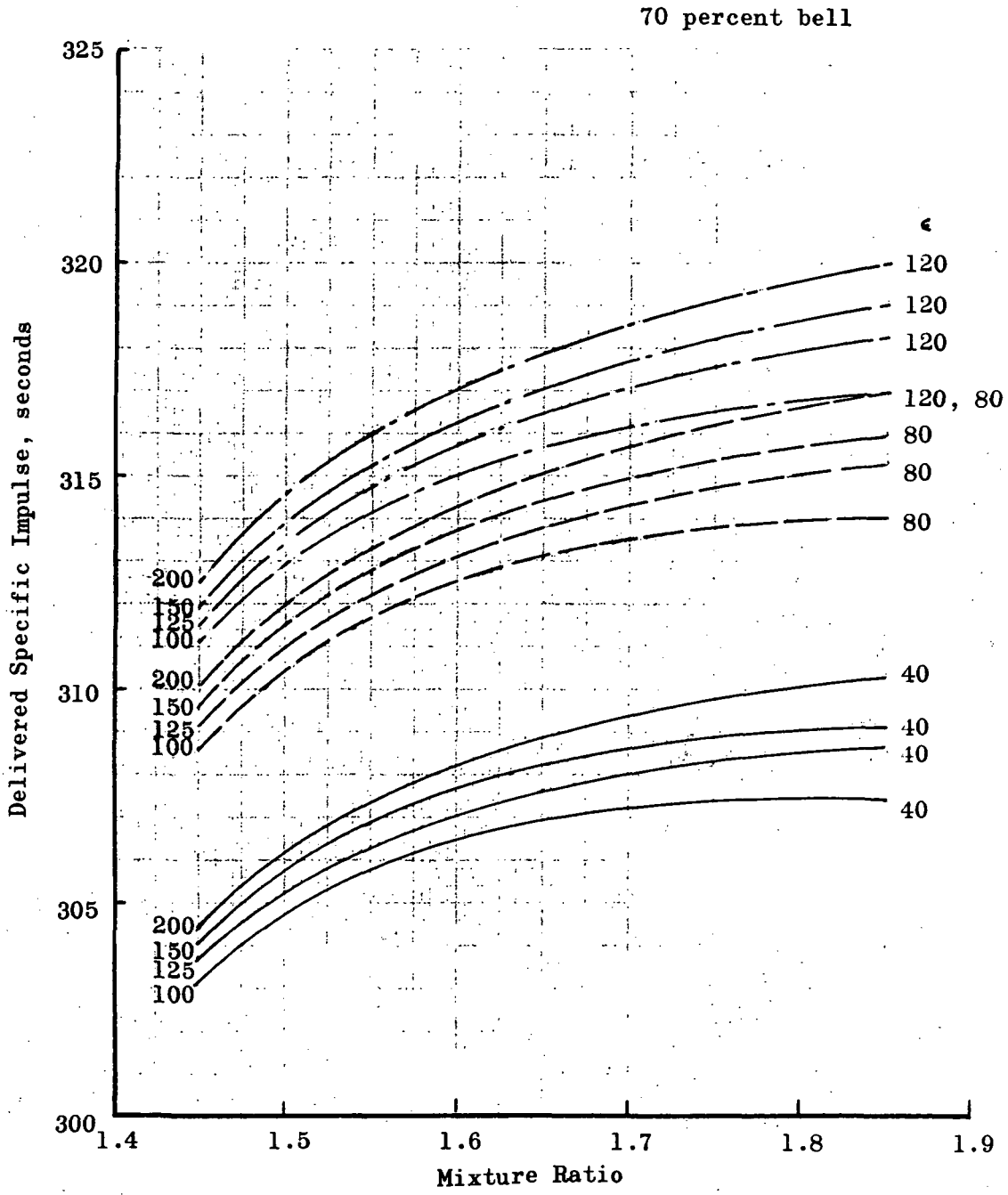


Figure 66. Delivered Performance for Regeneratively Cooled NTO/MMH Systems (6000 lb thrust)

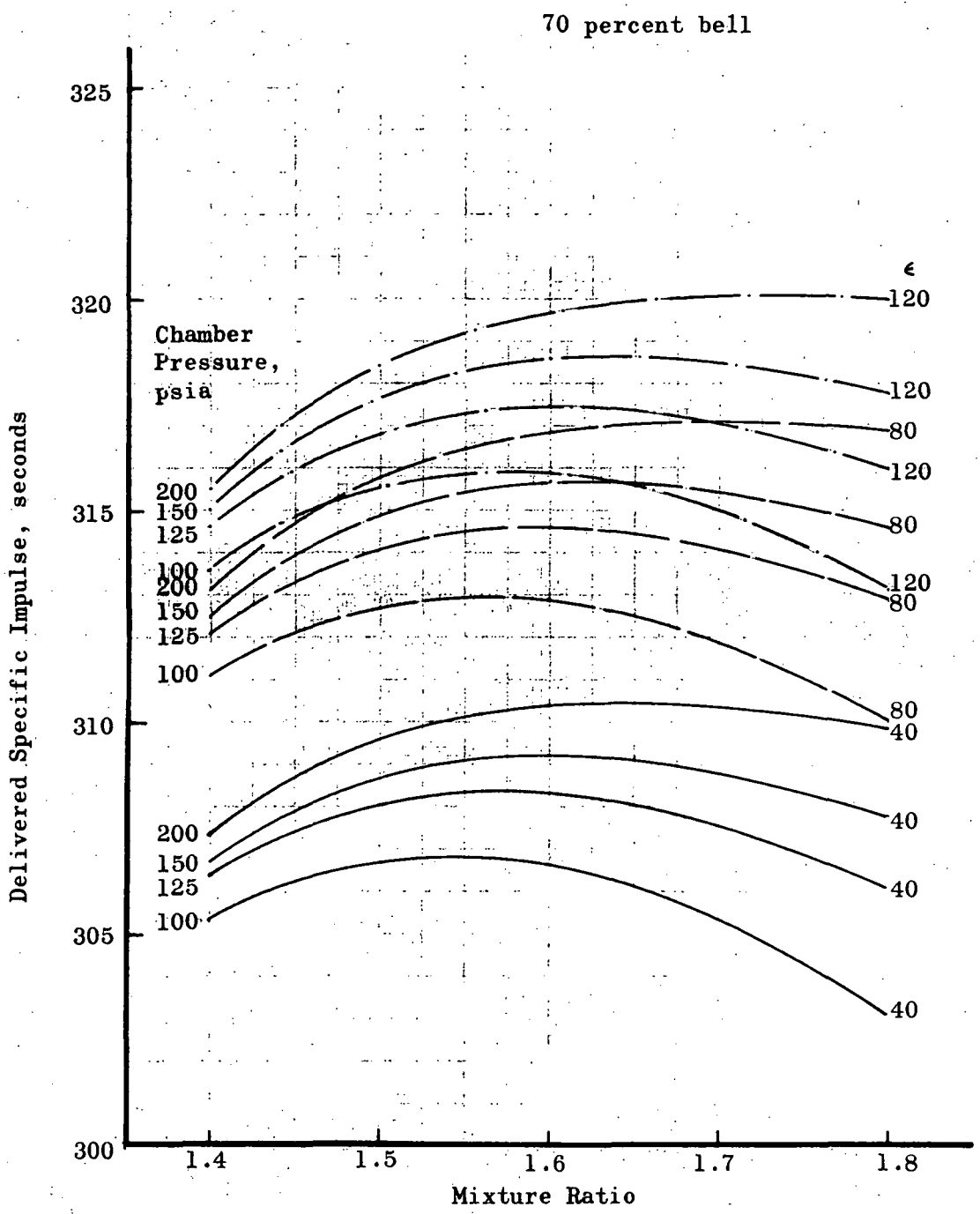


Figure 67. Delivered Performance for Regeneratively Cooled NTO/50-50 Systems (6000 lb thrust)

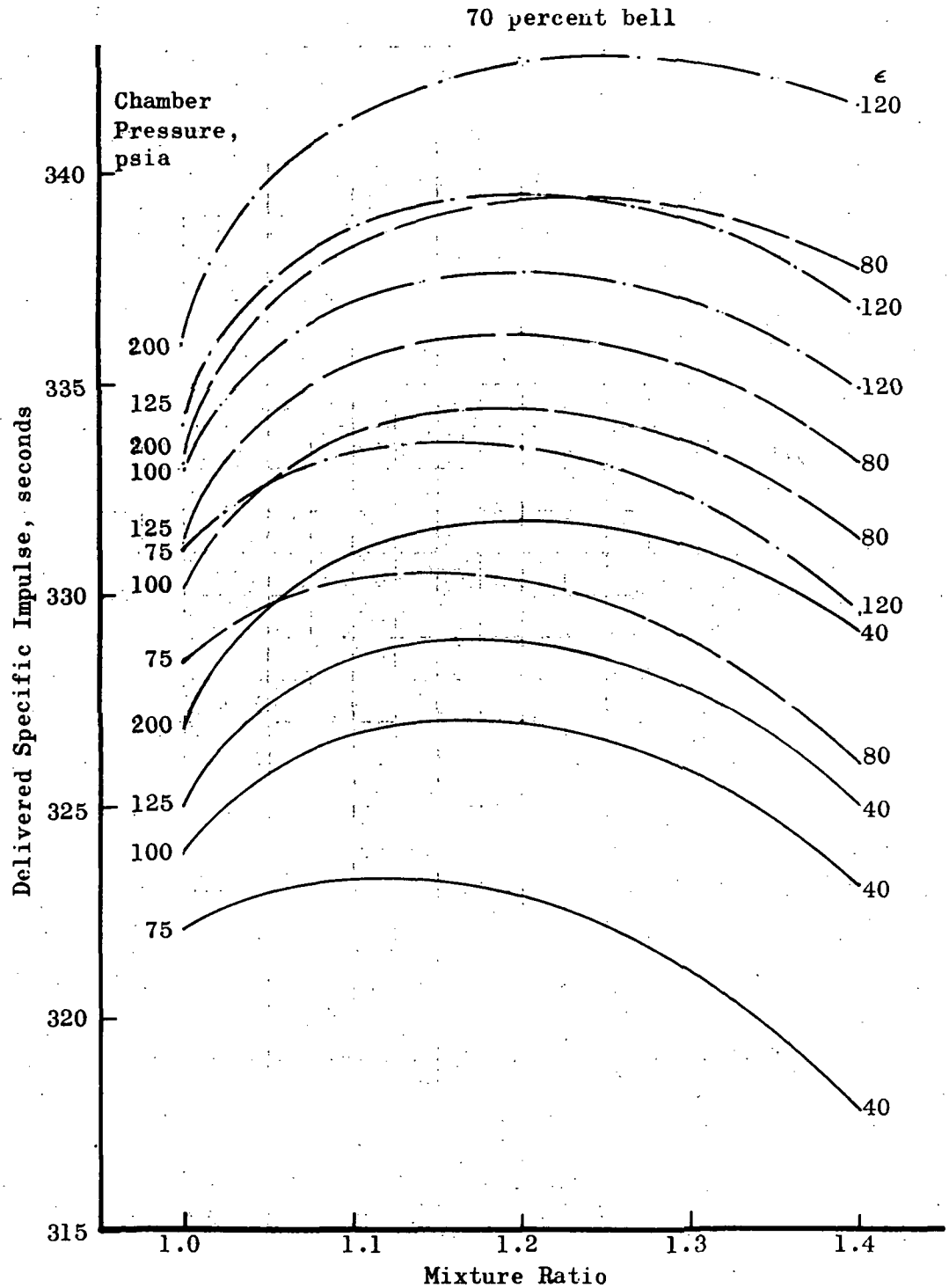


Figure 68. Delivered Performance for Regeneratively Cooled O_2/MMH Systems (6000 lb thrust)

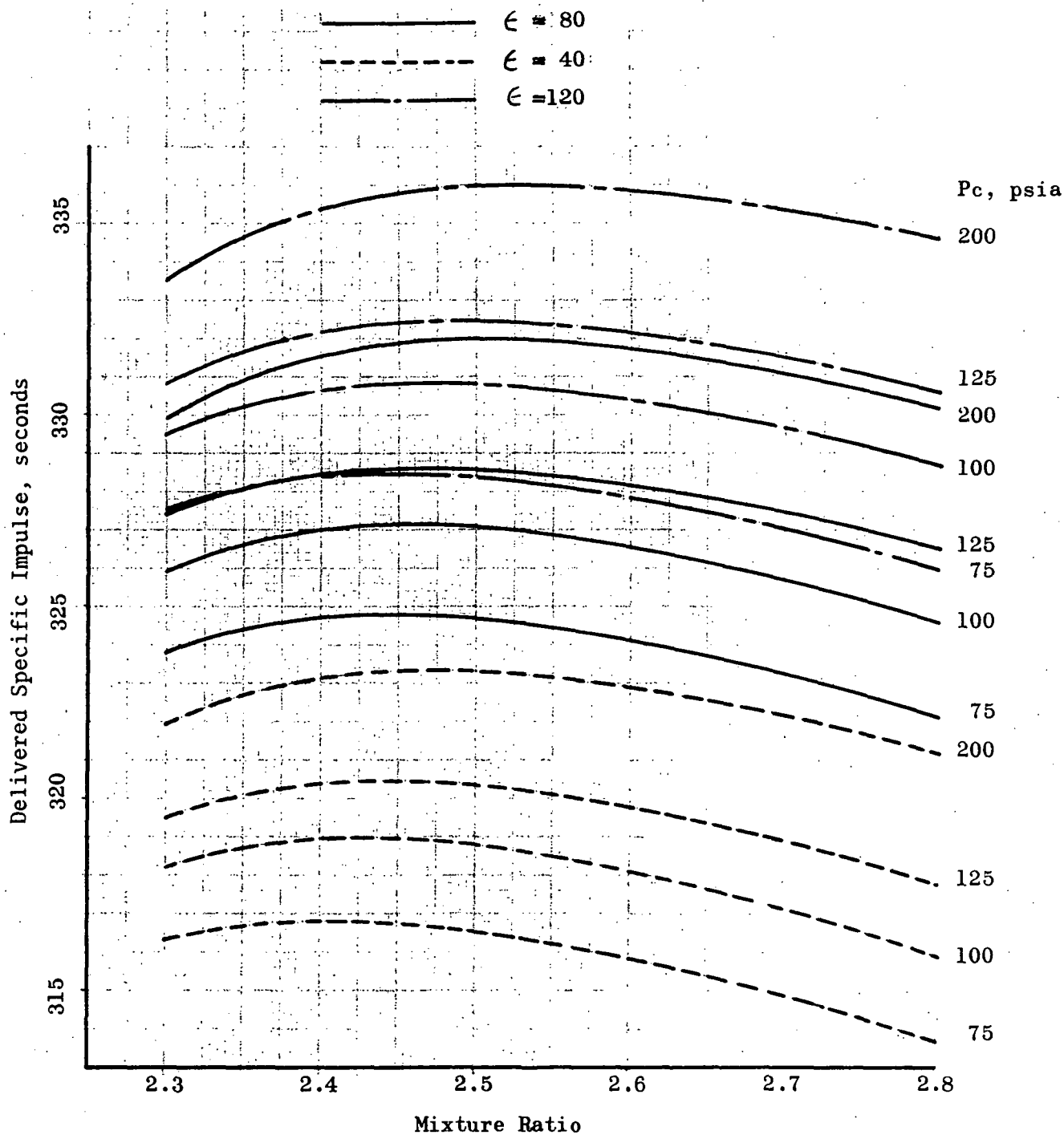


Figure 69. Delivered Performance for Regeneratively Cooled O_2/RP-1 Systems (6000 lb Thrust)

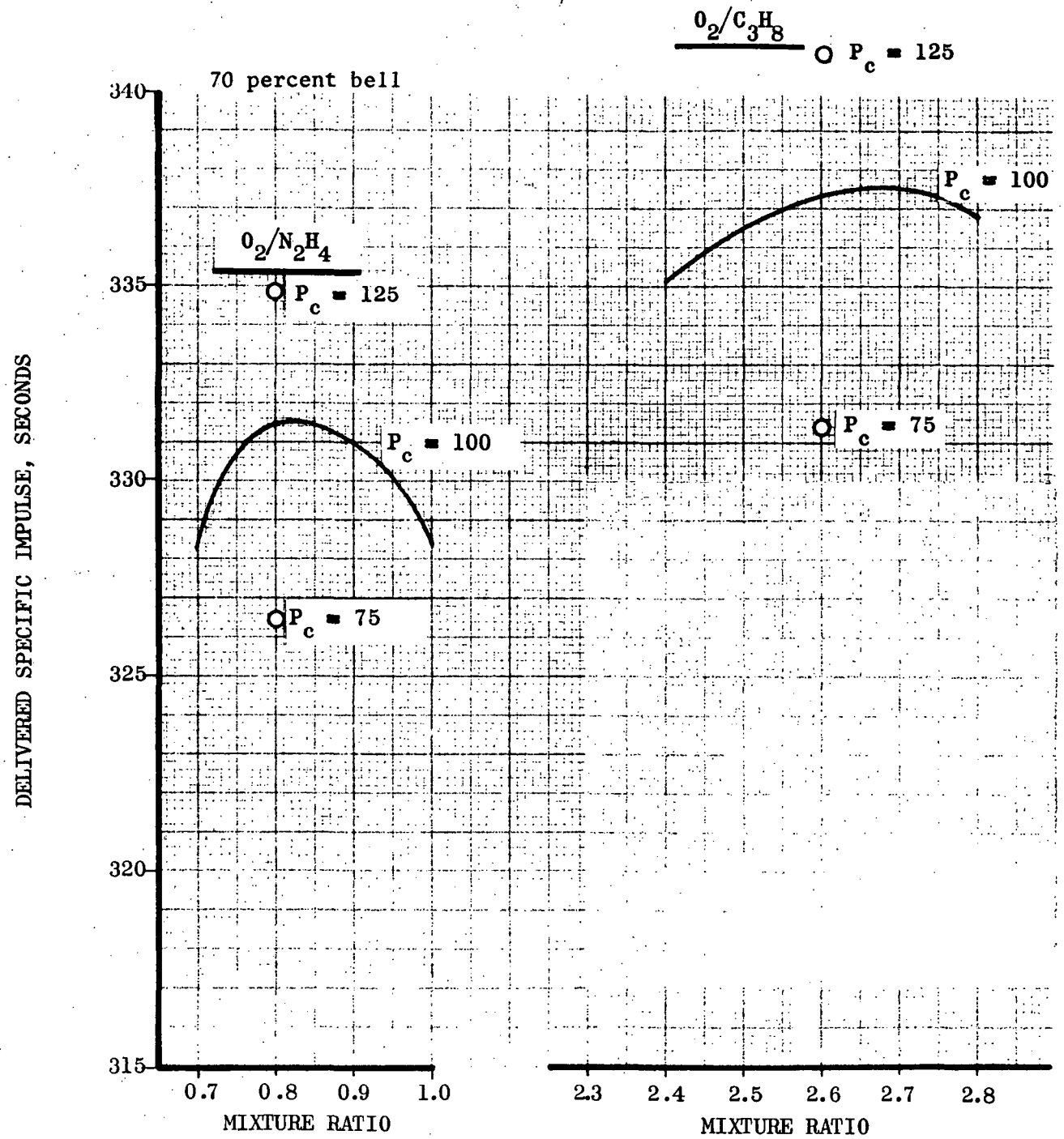


Figure 70. Delivered Performance for Regenerative Cooled O_2/N_2H_4 and O_2/C_3H_8 Systems (6000 lb thrust)

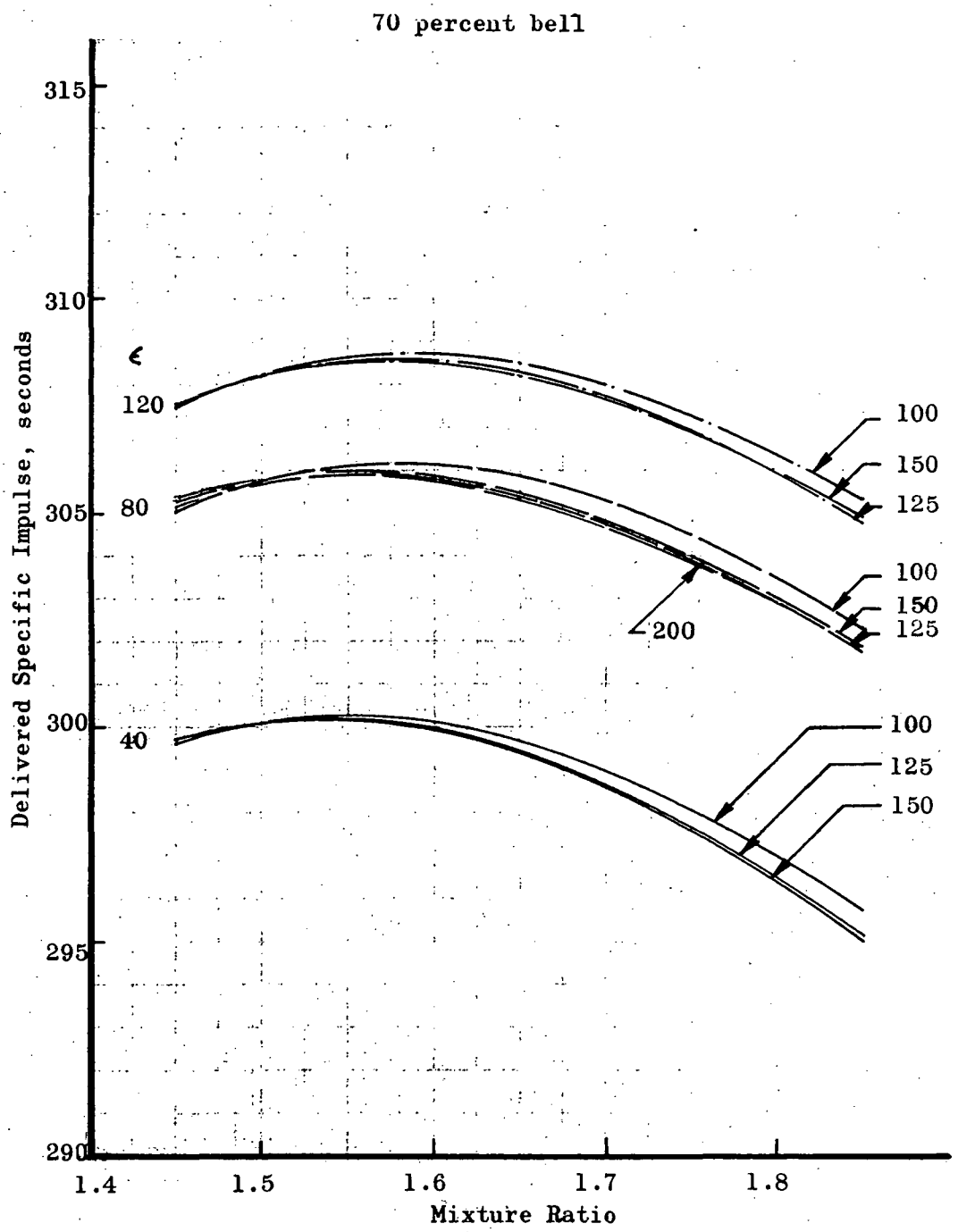


Figure 71. Delivered Performance for Dump/Film Cooled NTO/MMH Systems,(6000 lb thrust)

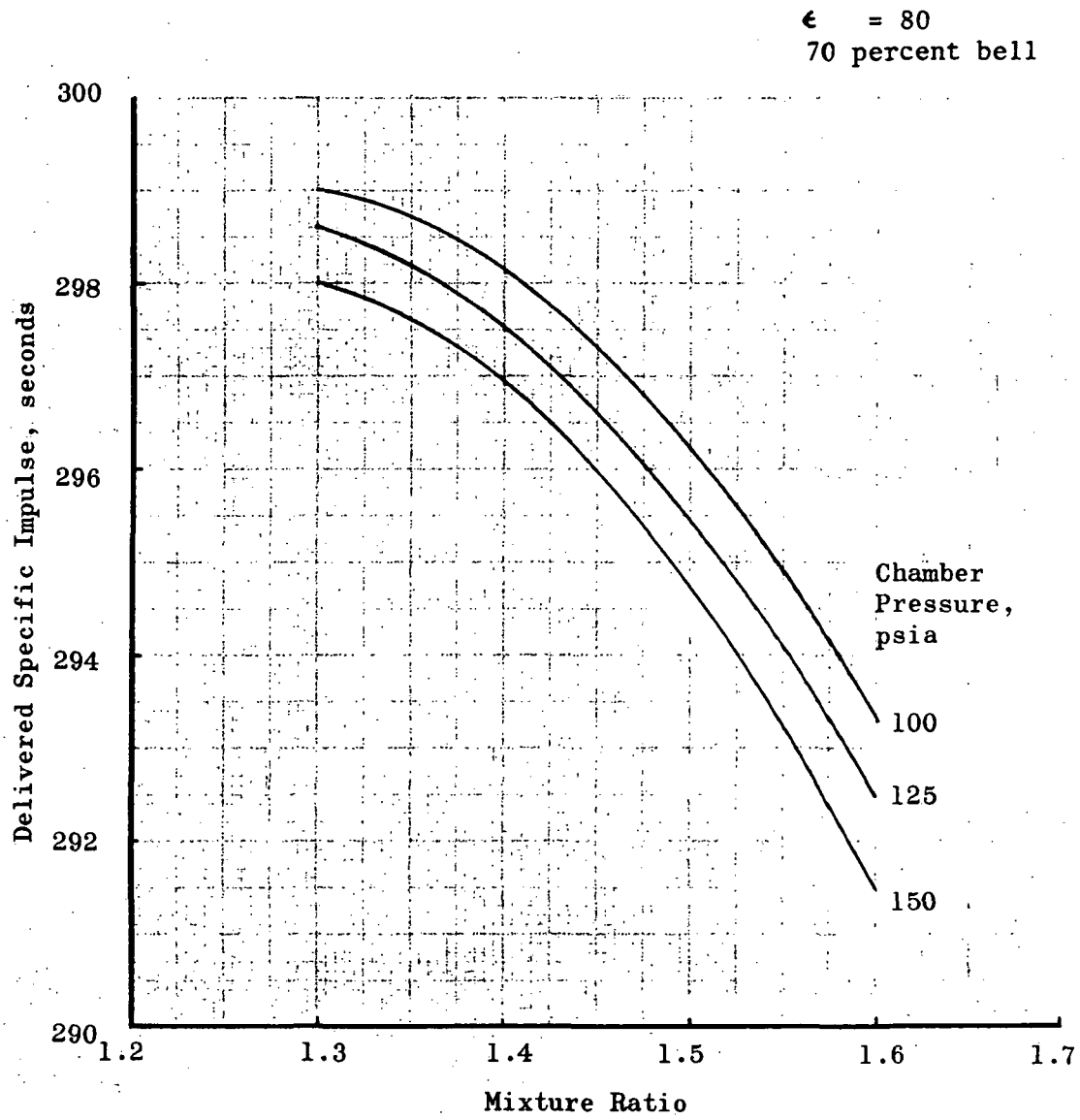


Figure 72. Delivered Performance for Dump/Film Cooled NTO/50-50 System (6000 lb thrust)

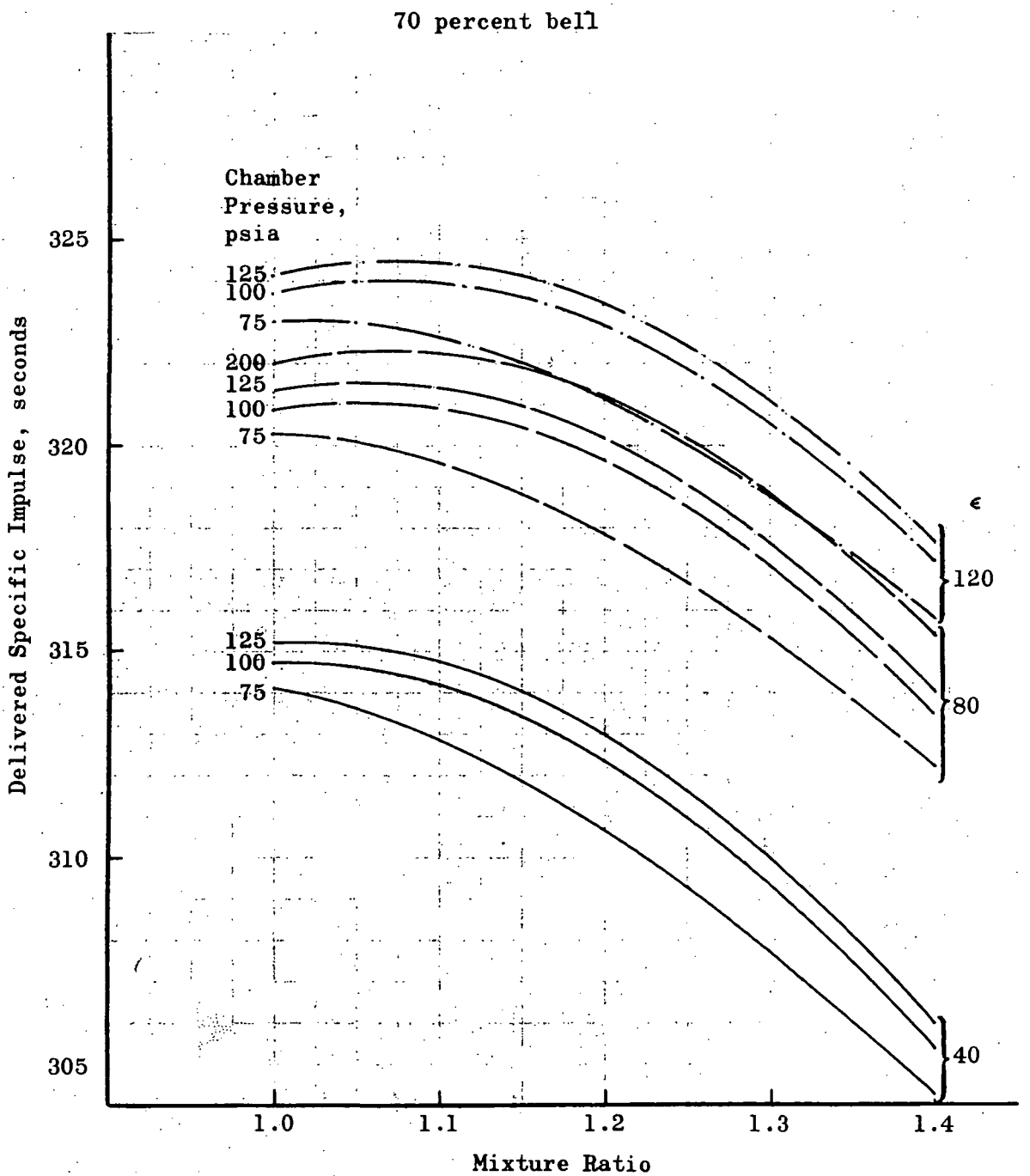


Figure 73. Delivered Performance for Dump/Film Cooled O_2/MMH Systems (6000 lb thrust)

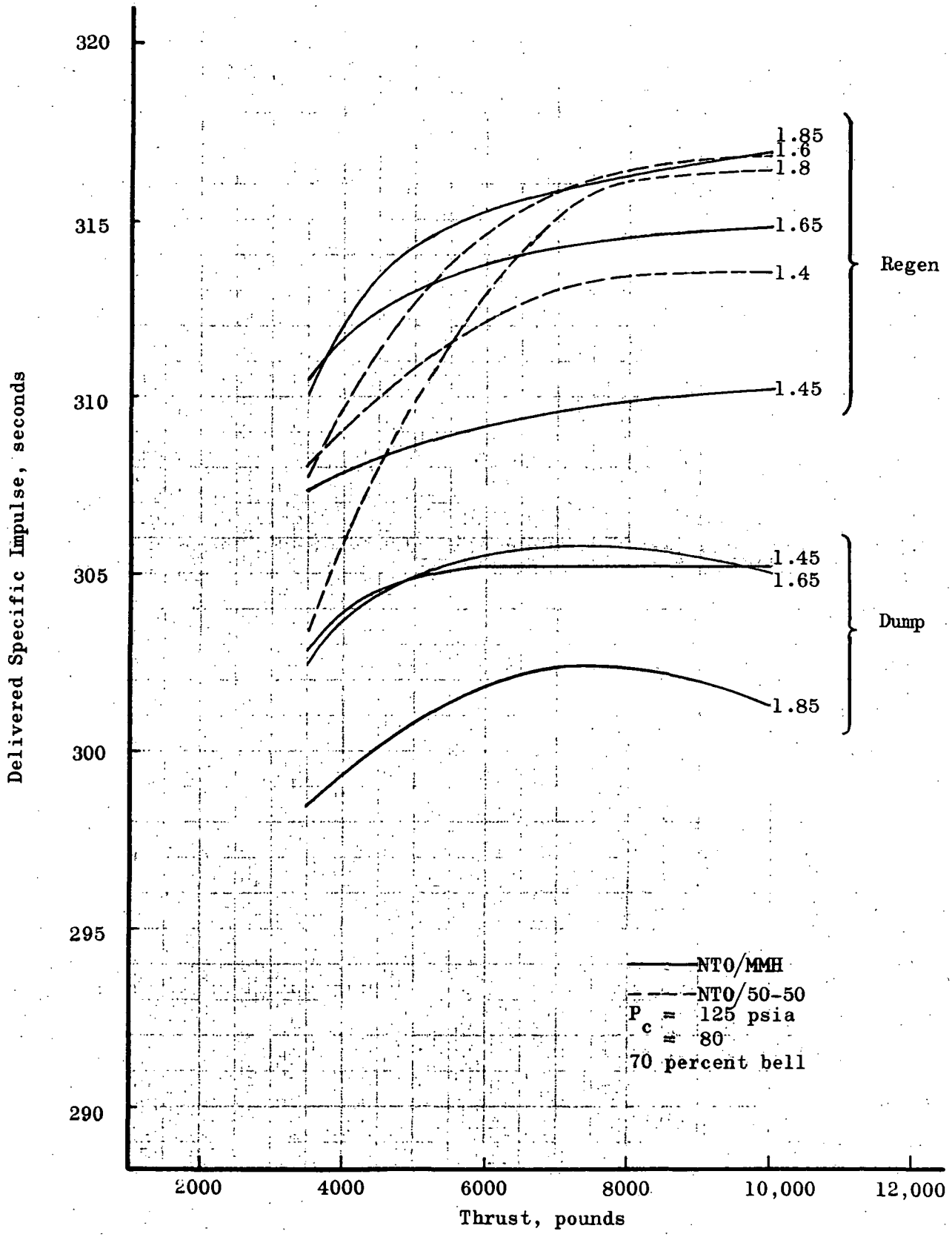


Figure 74. Delivered Performance vs Thrust: NTO/MMH and NTO/50-50

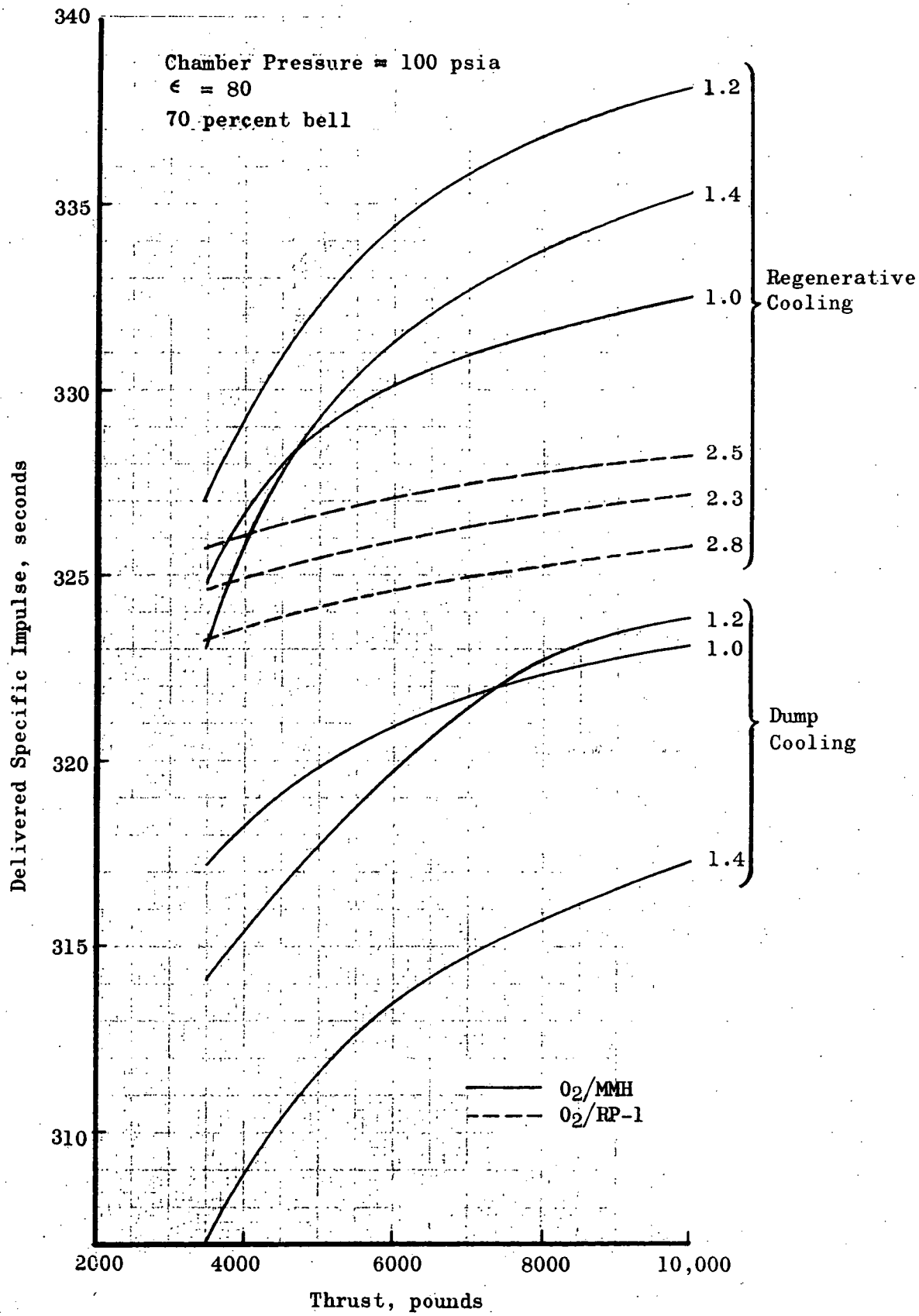


Figure 75. Delivered Performance vs Thrust O_2/MMH and O_2/RP-1

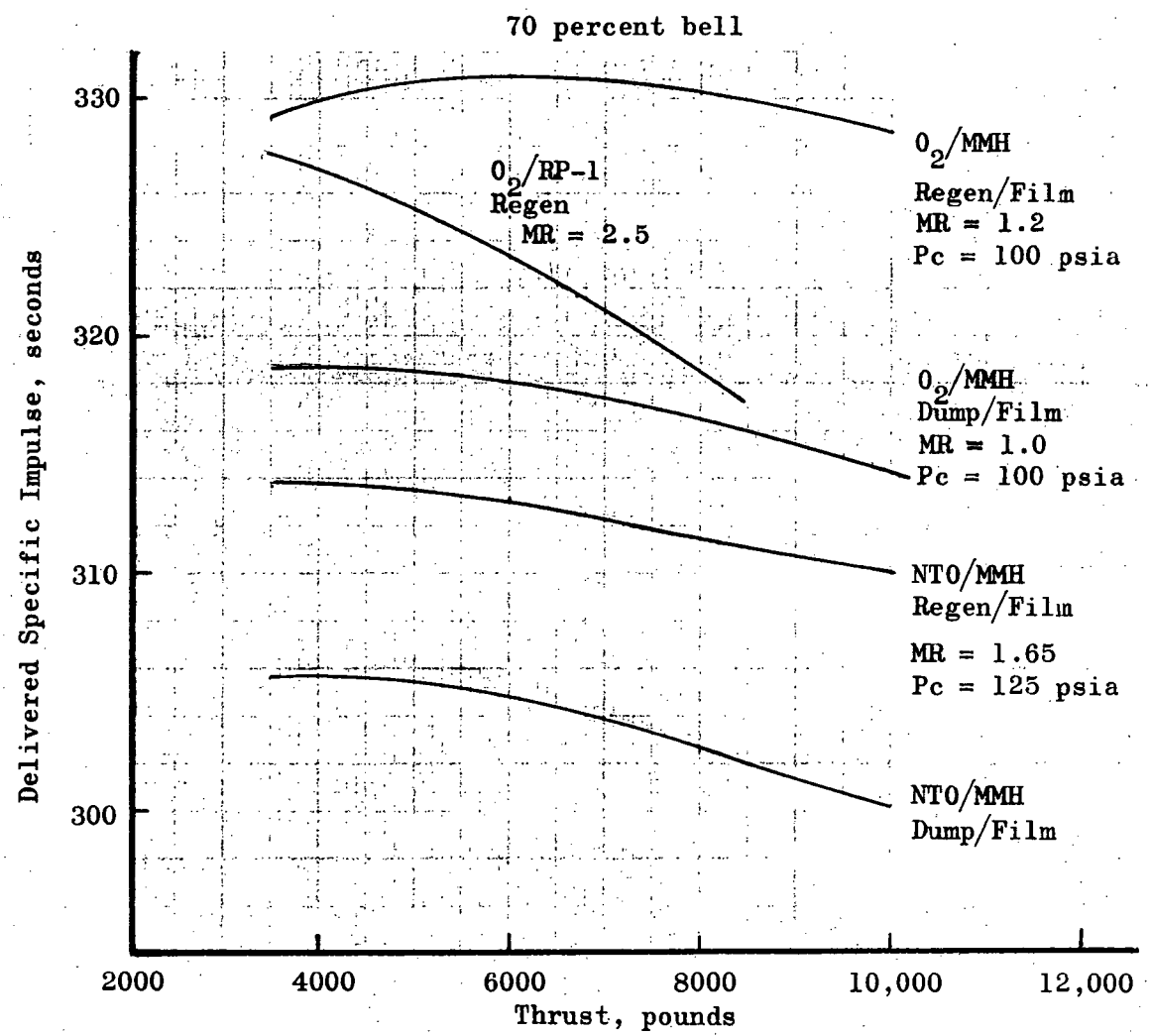


Figure 76. Delivered Specific Impulse for 50-Inch Diameter Systems.

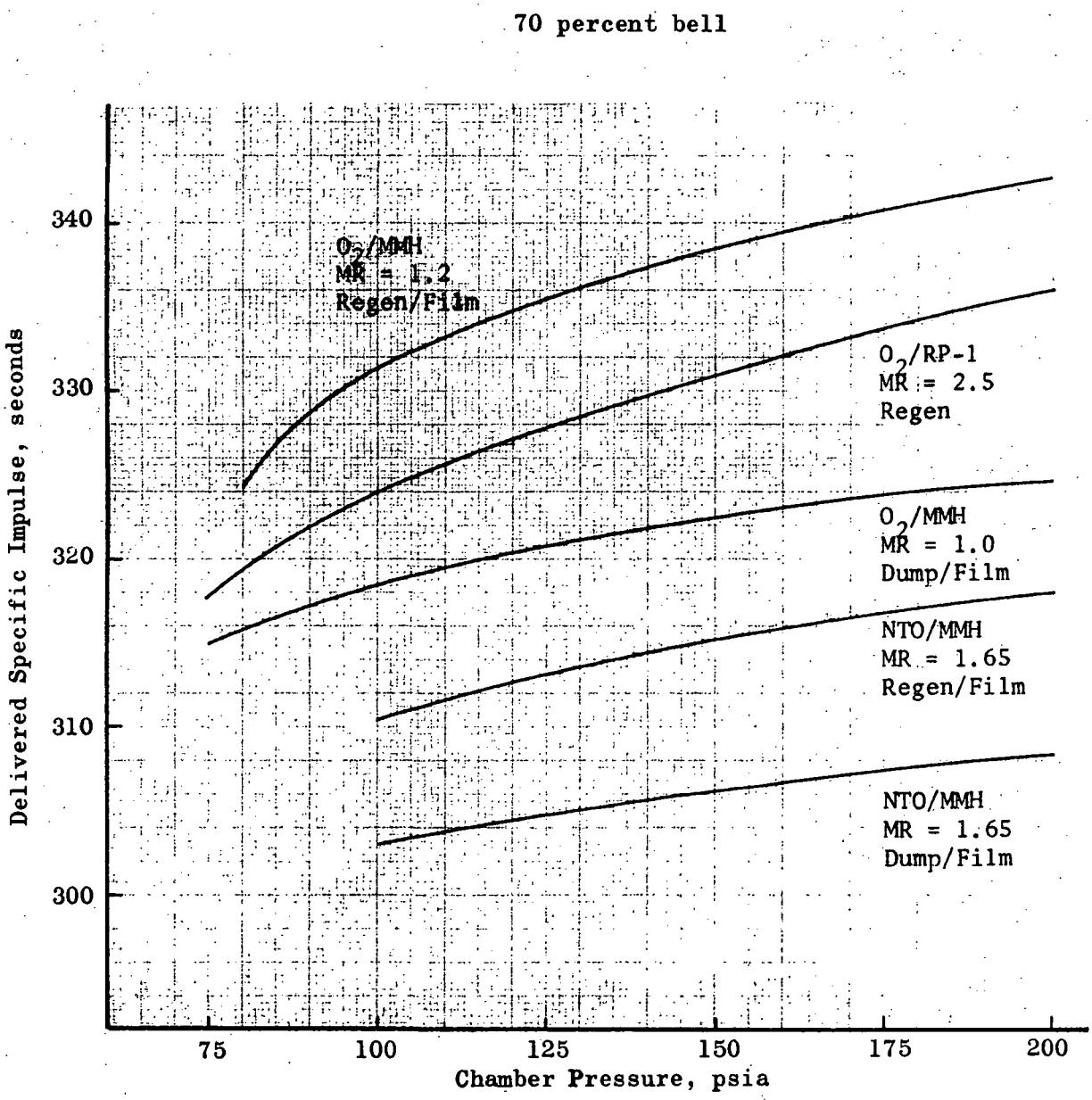


Figure 77. Effect of Chamber Pressure on Performance in Constant Diameter 6000-Pound Thrust System.

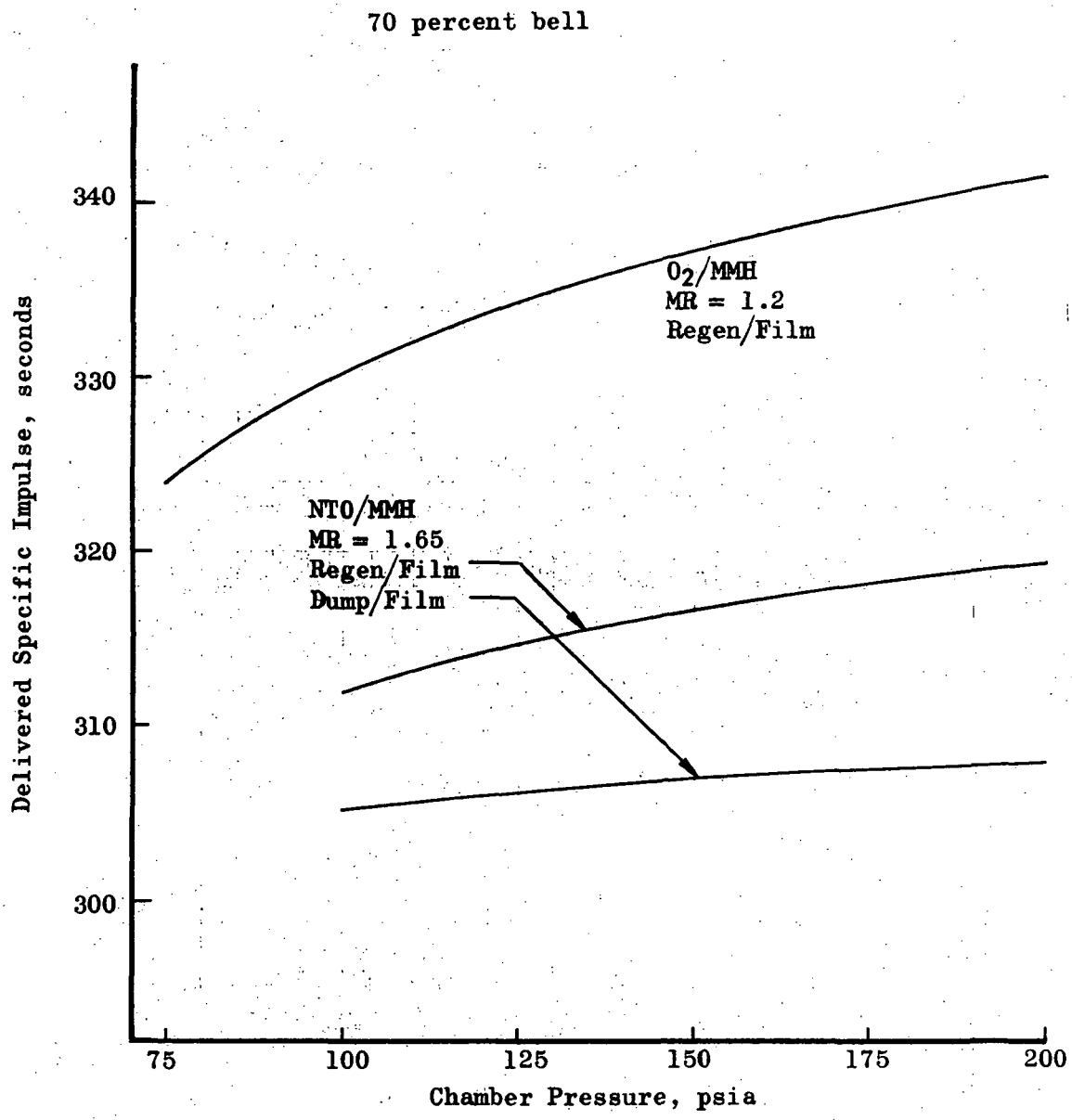


Figure 78. Effect of Chamber Pressure on Performance in Constant Diameter 4000-pound Thrust System

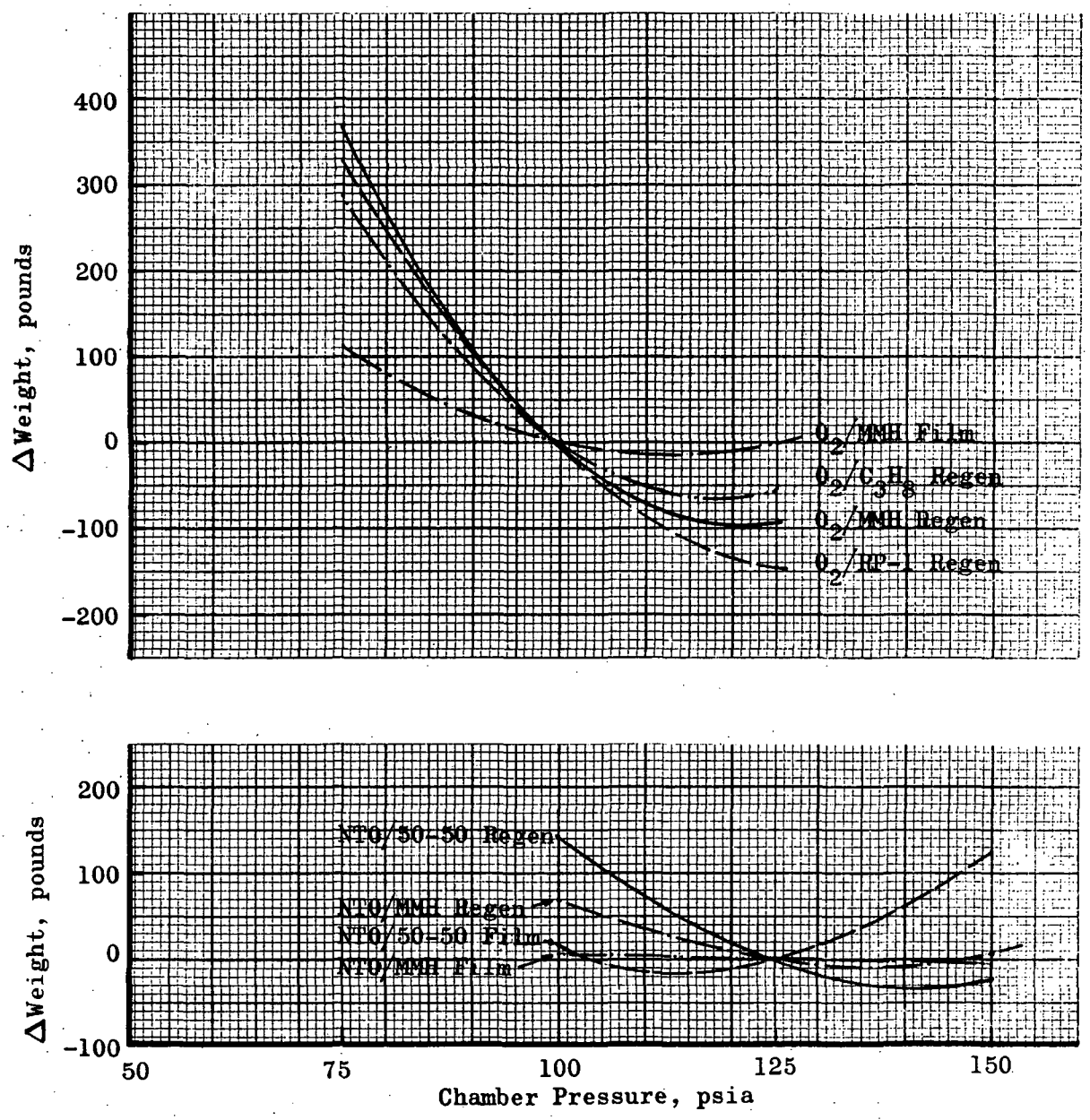


Figure 79. Chamber Pressure Effect on Total OMS Weight

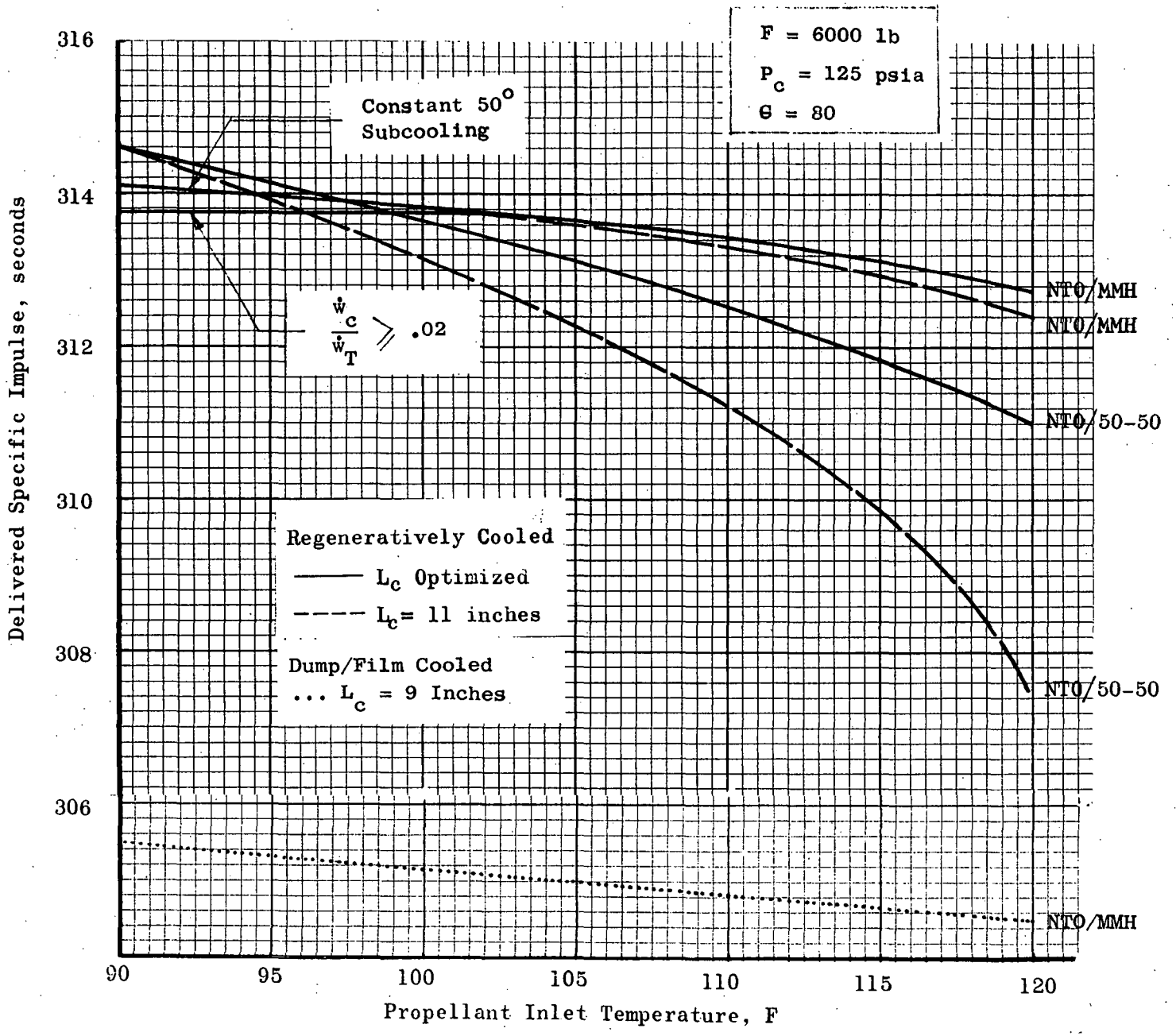


Figure 80. Performance Penalties at Design Point for Maximum Propellant Inlet Temperature

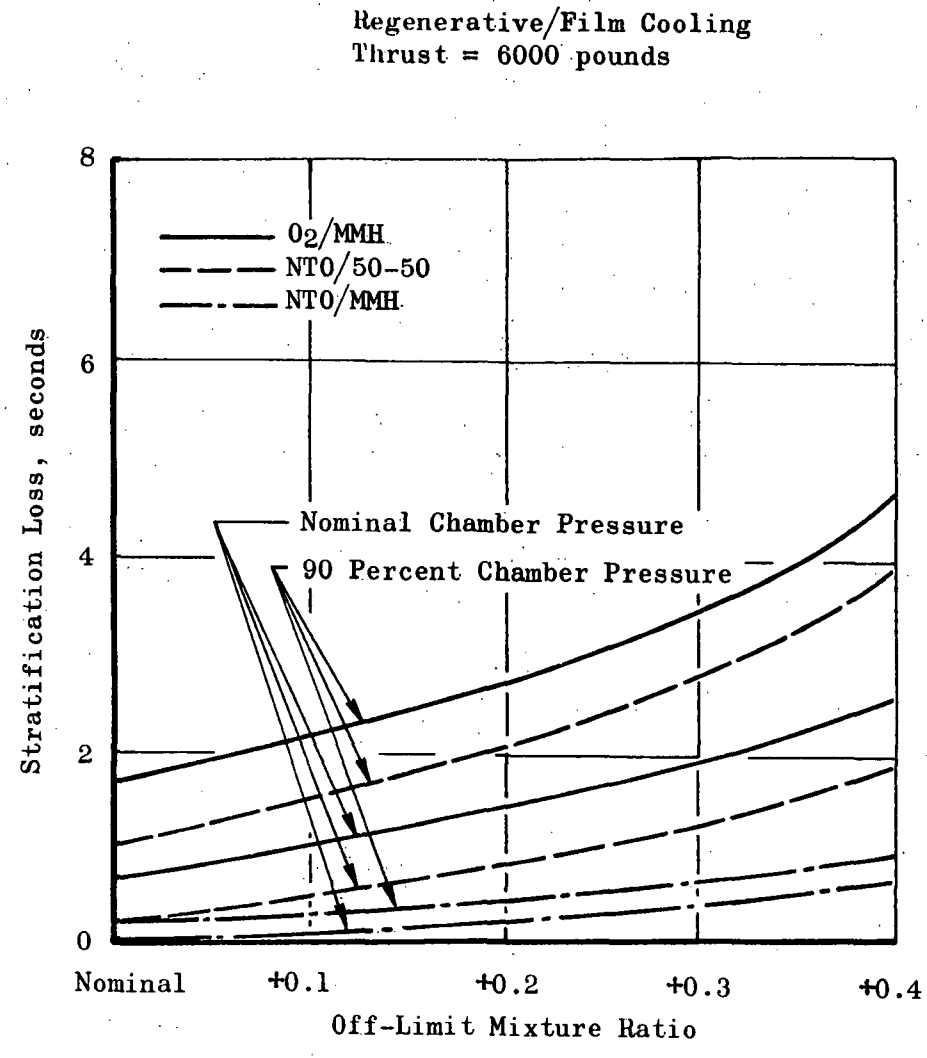


Figure 81. Performance Penalties at Design Point for Off-Limit Capabilities

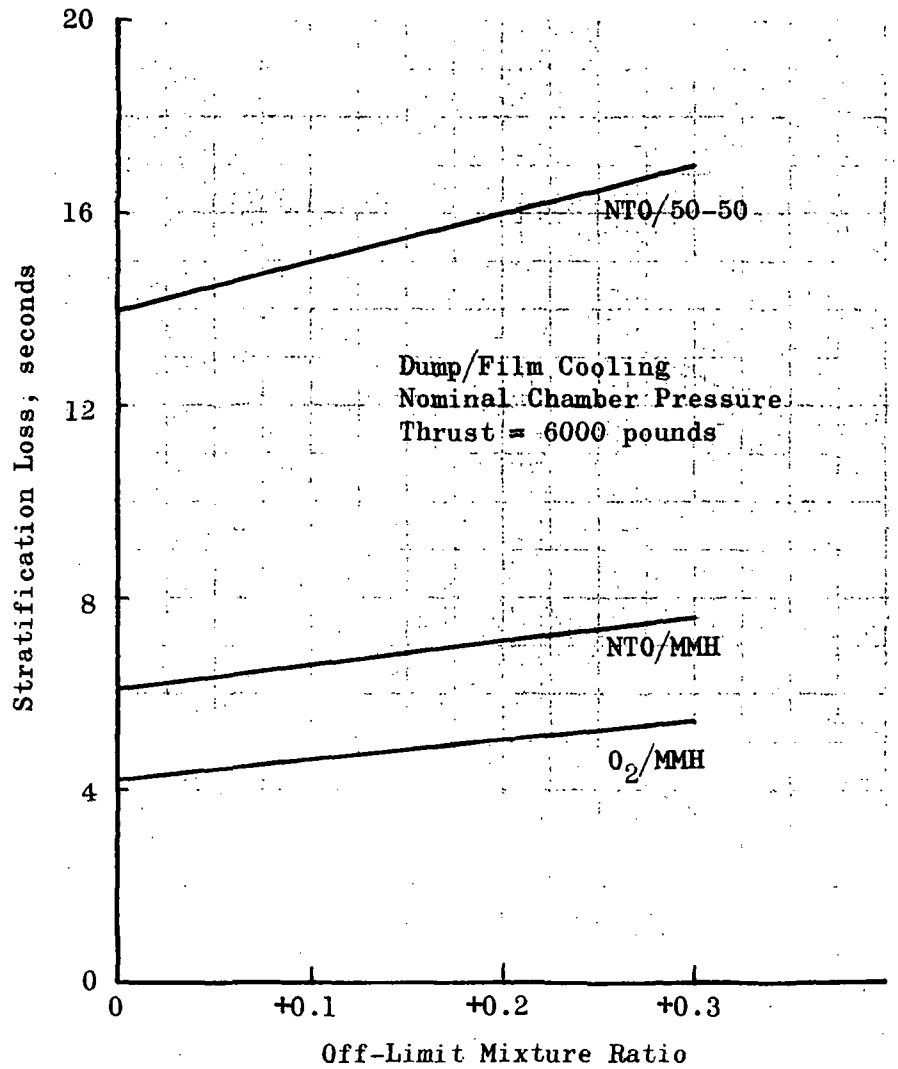


Figure 82 Performance Penalties at Design Point for Off-Limit Capabilities

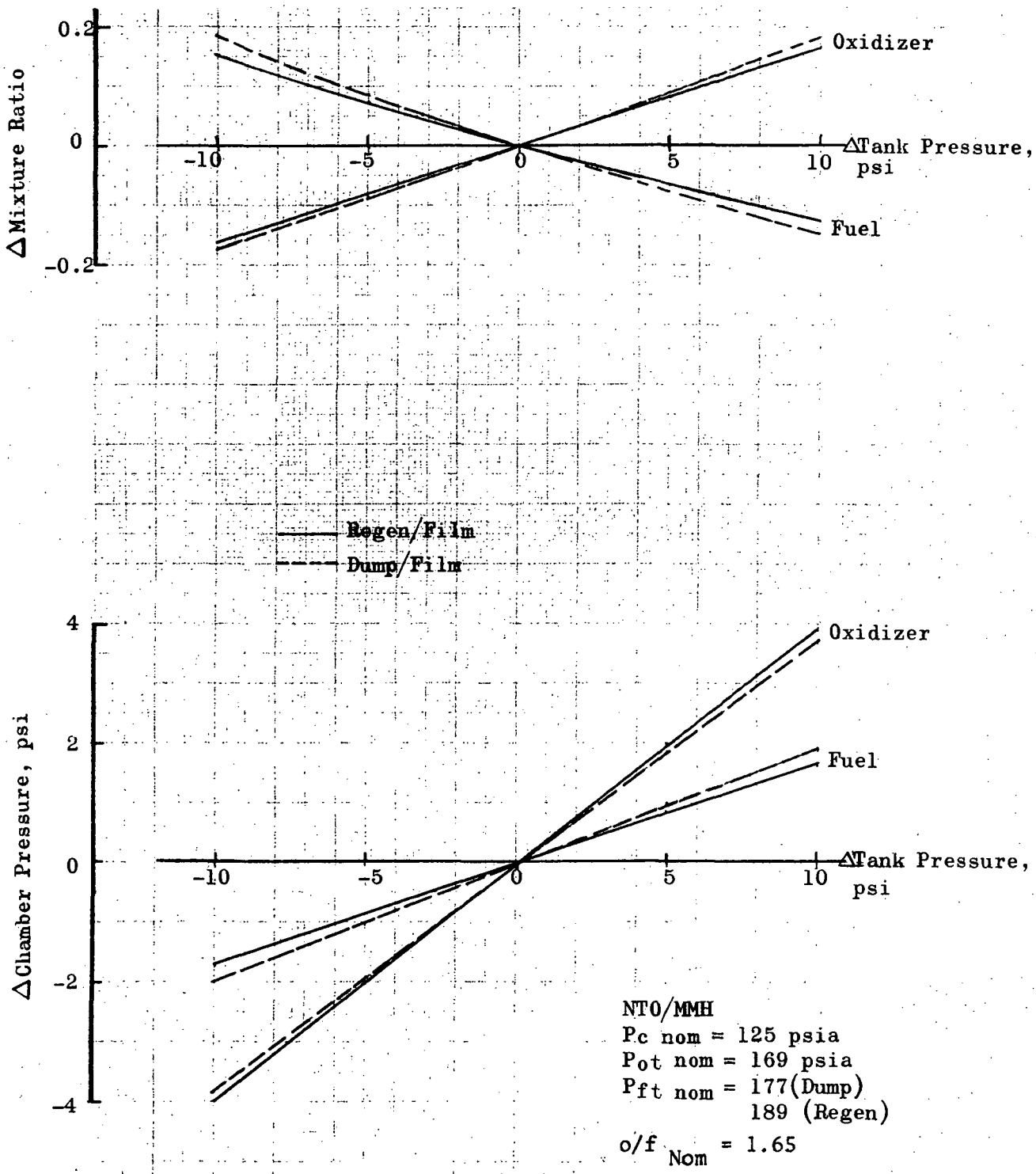


Figure 83. Inlet Pressure Sensitivity: NTO/MMH

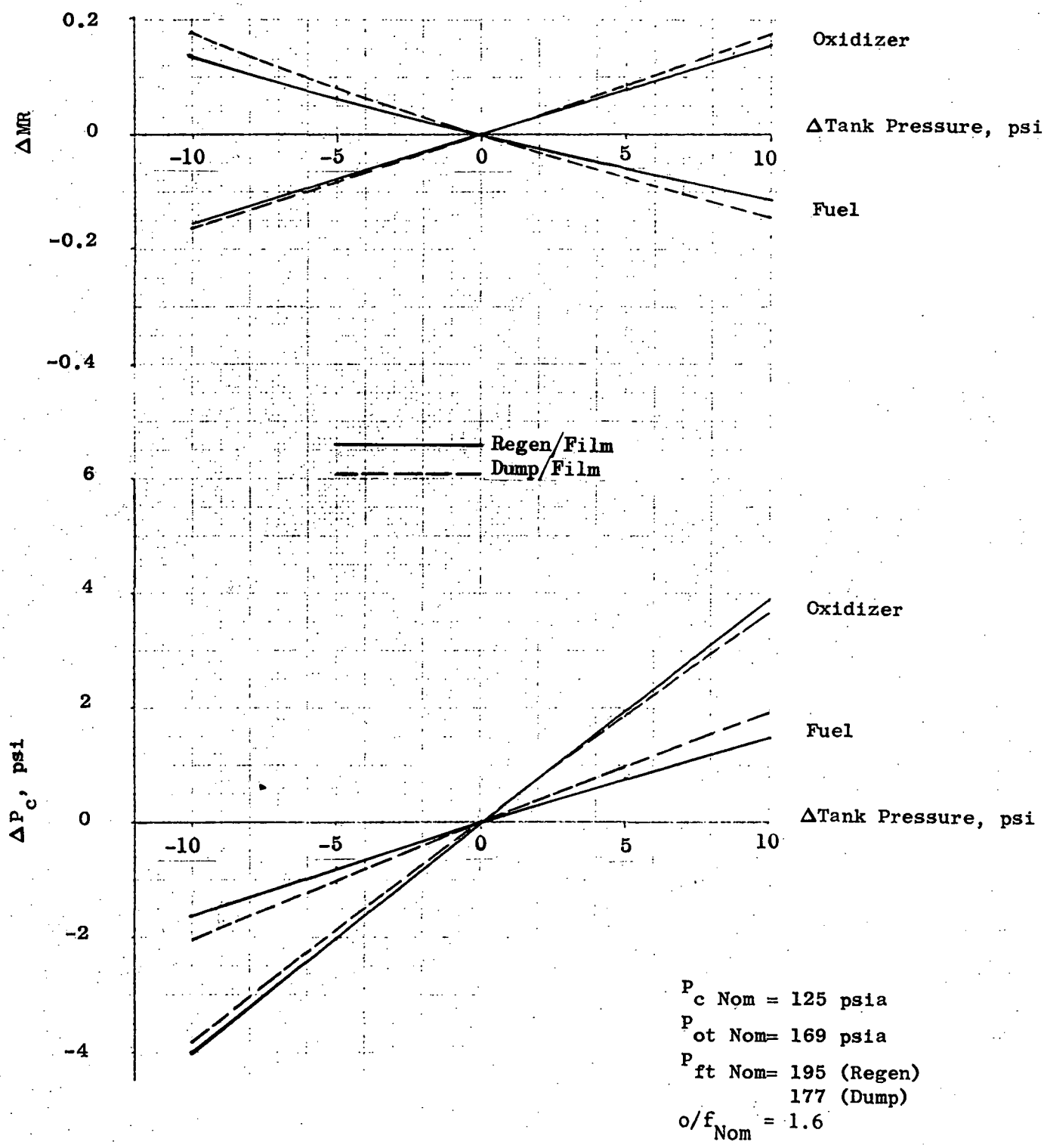


Figure 84. Inlet Pressure Sensitivity: NTO/50-50

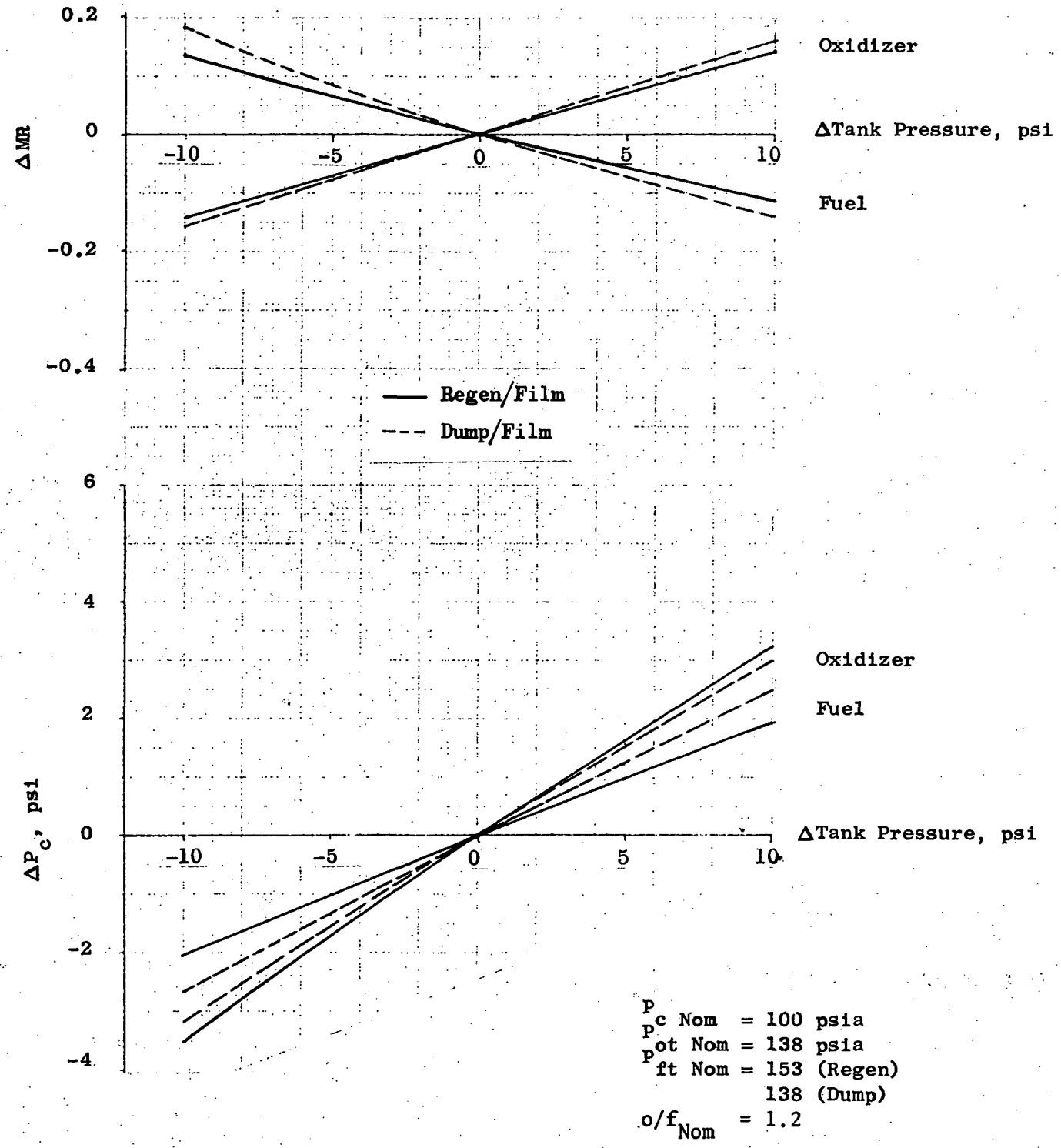


Figure 85. Inlet Pressure Sensitivity: LOX/MMH

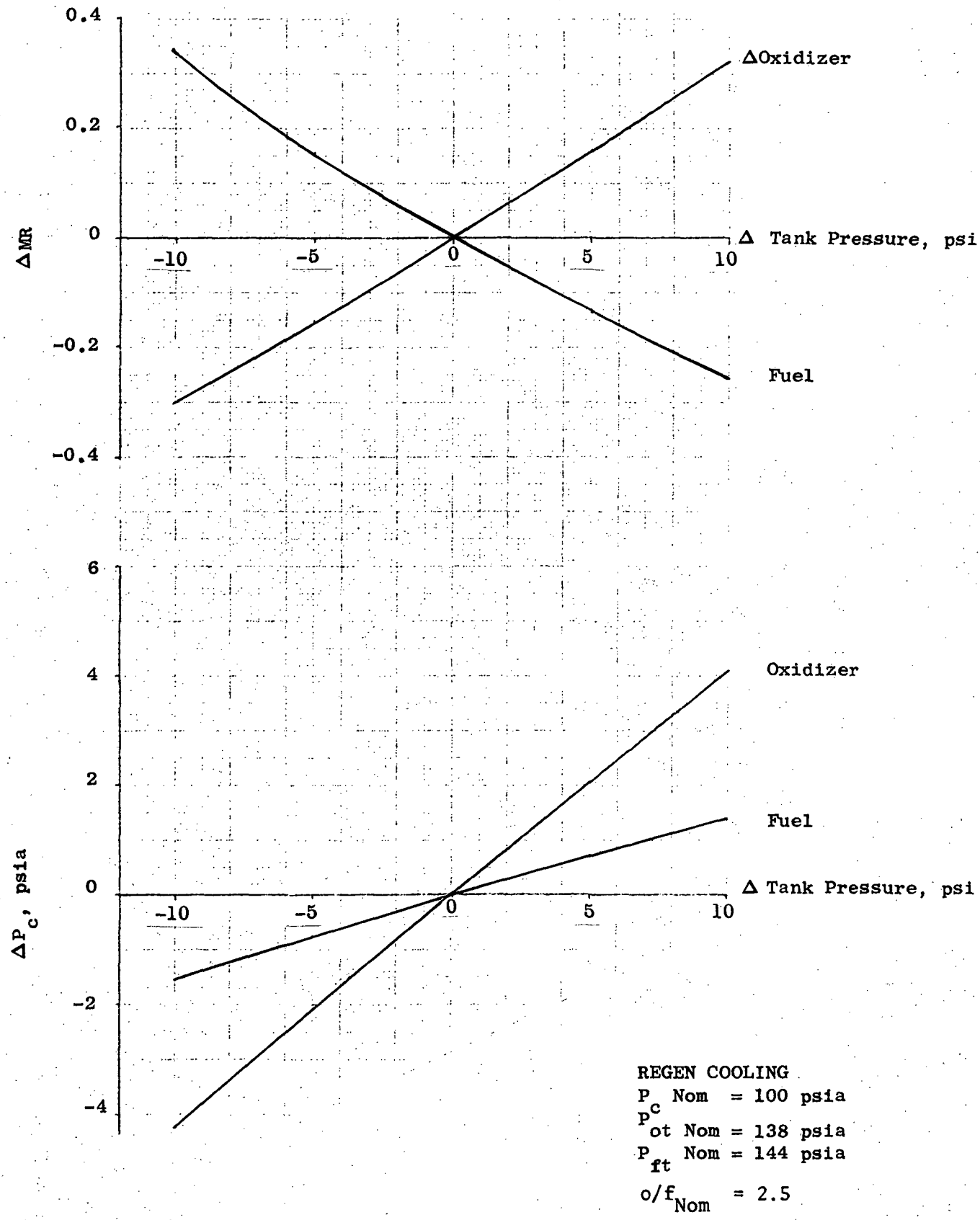


Figure 86. Inlet Pressure Sensitivity: LOX/RP-1

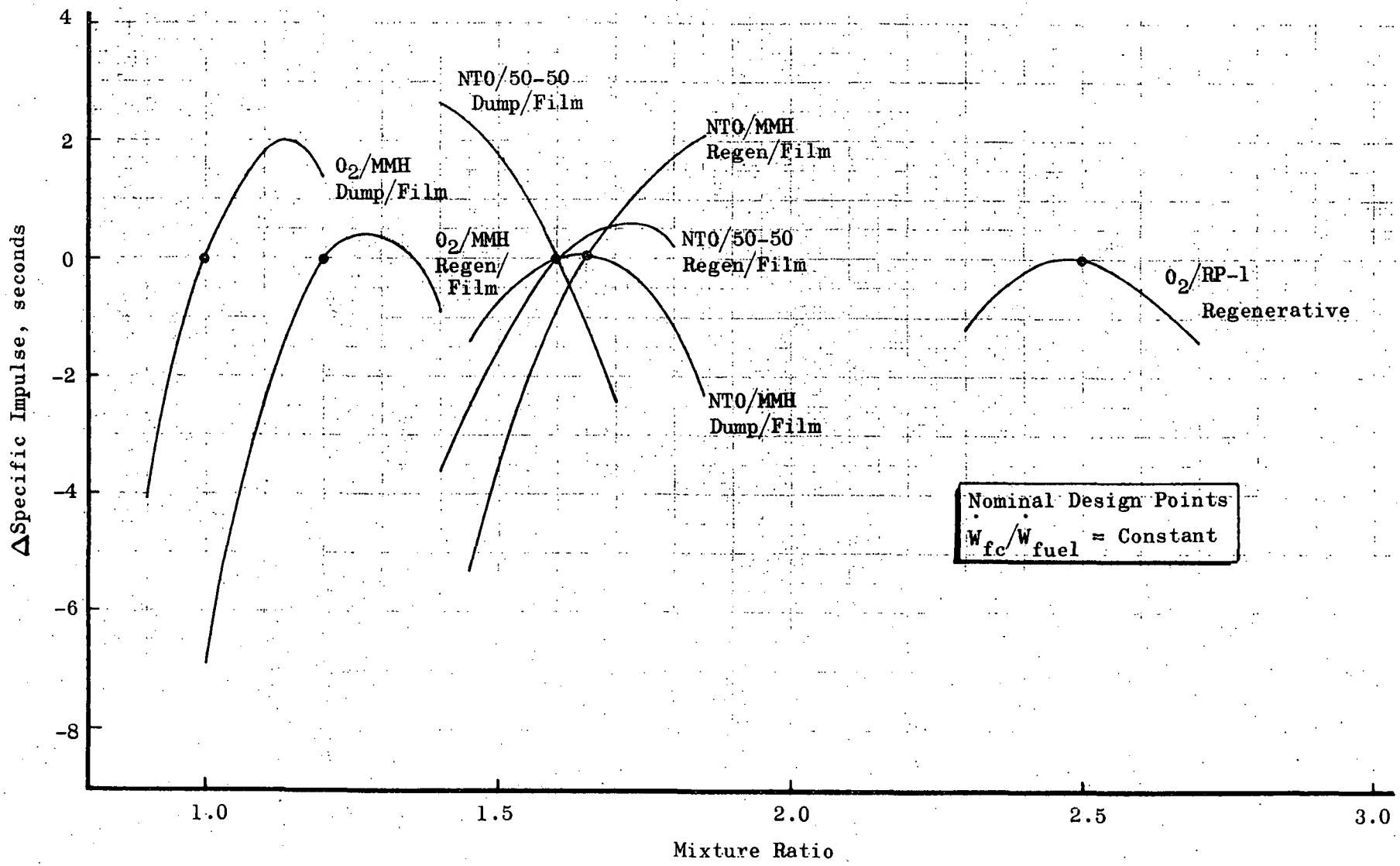


Figure 87. Operating Sensitivity to Mixture Ratio Variations

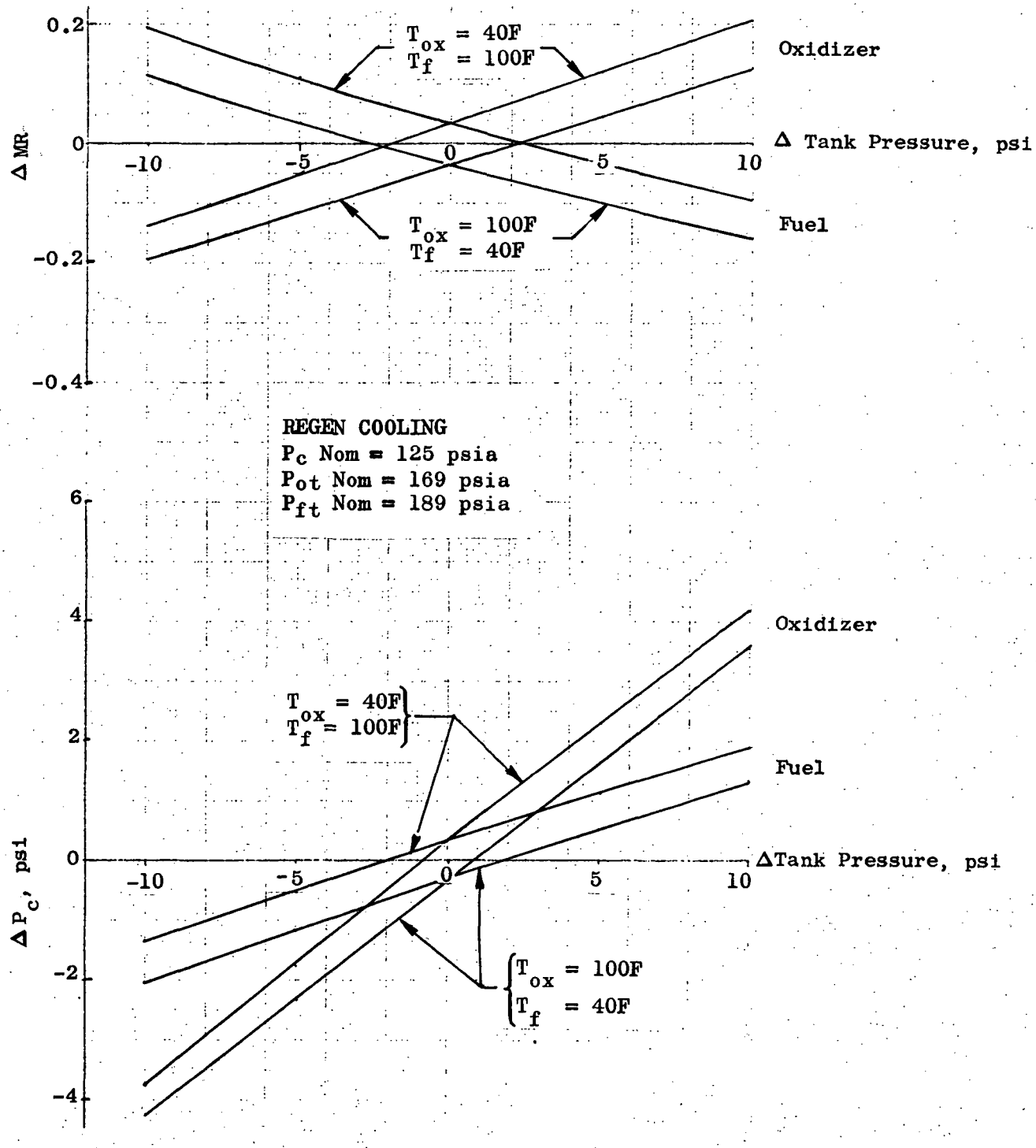
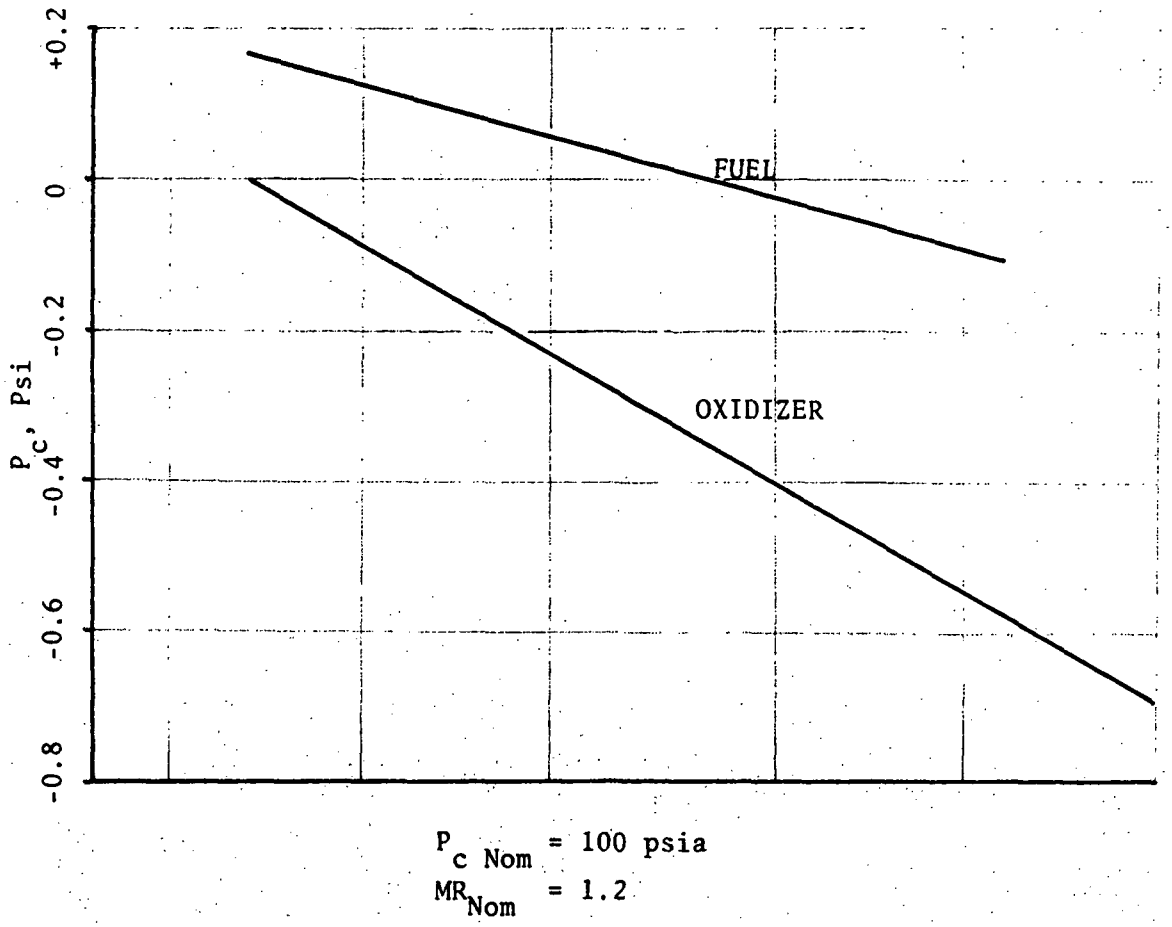


Figure 88. Pressure and Temperature Sensitivity: NTO/MMH



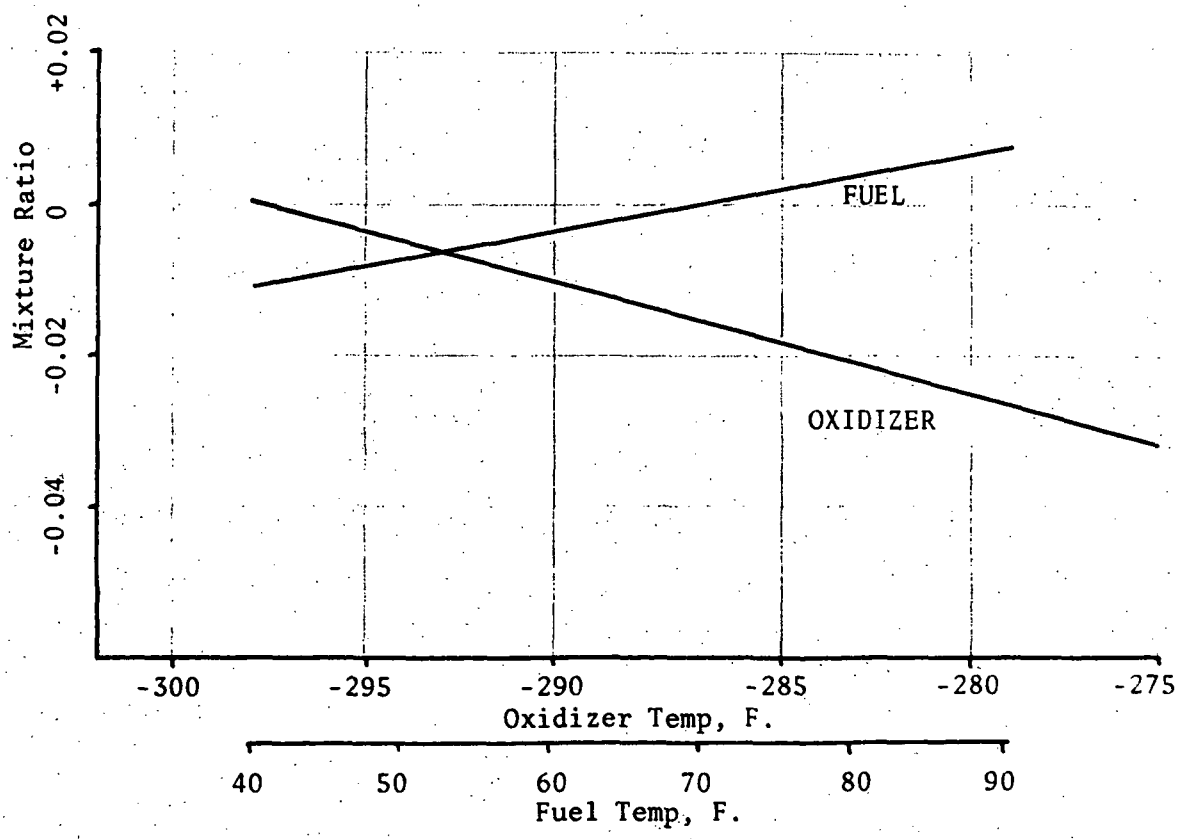


Figure 89. Inlet Temperature Sensitivity in O_2/MMH or O_2/50-50 Regenerative Cooled System

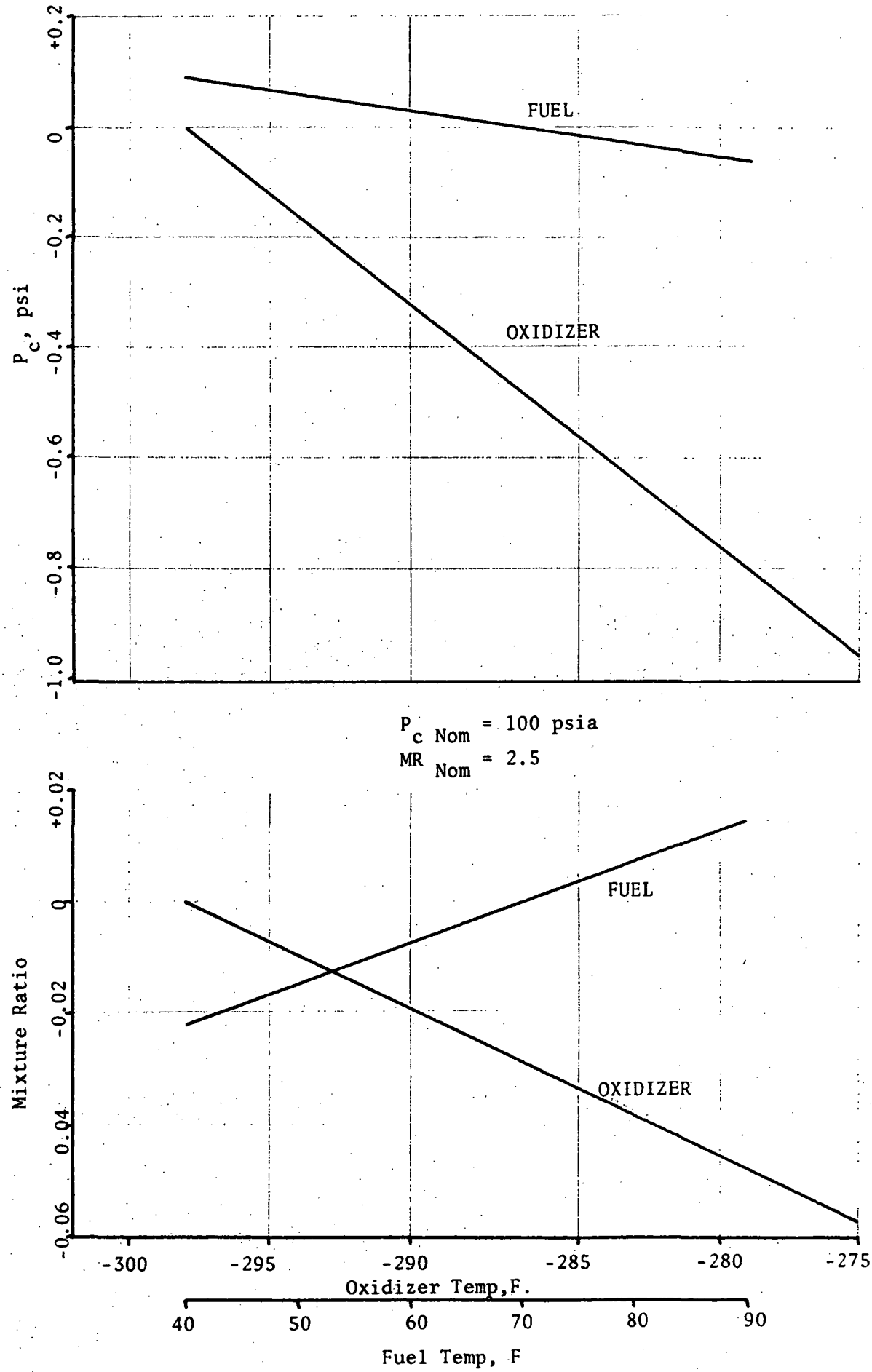


Figure 90. Inlet Temperature Sensitivity in O_2/RP-1 Regenerative Cooled System

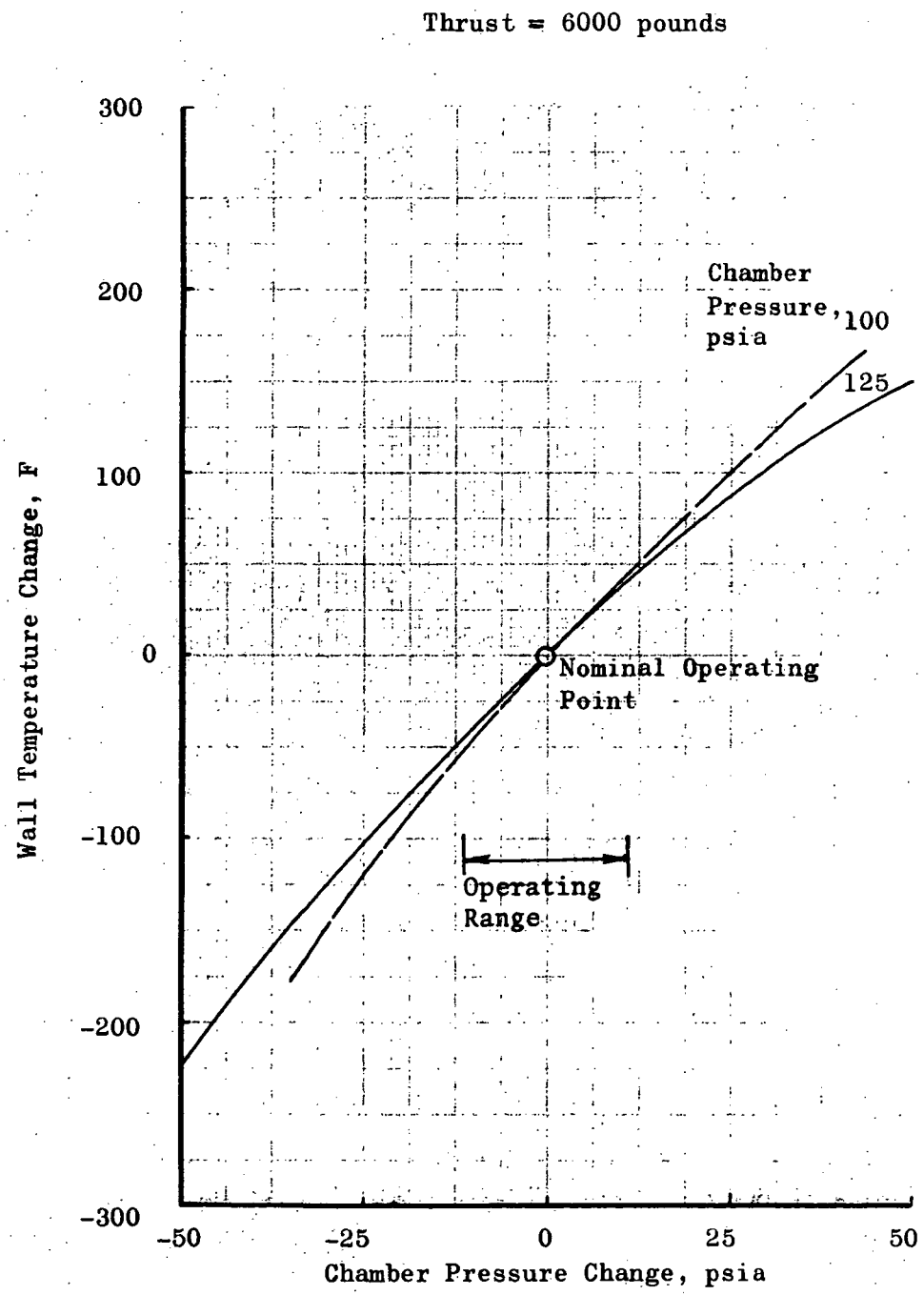


Figure 91. Sensitivity of Regenerative Cooled Engine Radiation Nozzle Maximum Temperature to Chamber Pressure Variations.

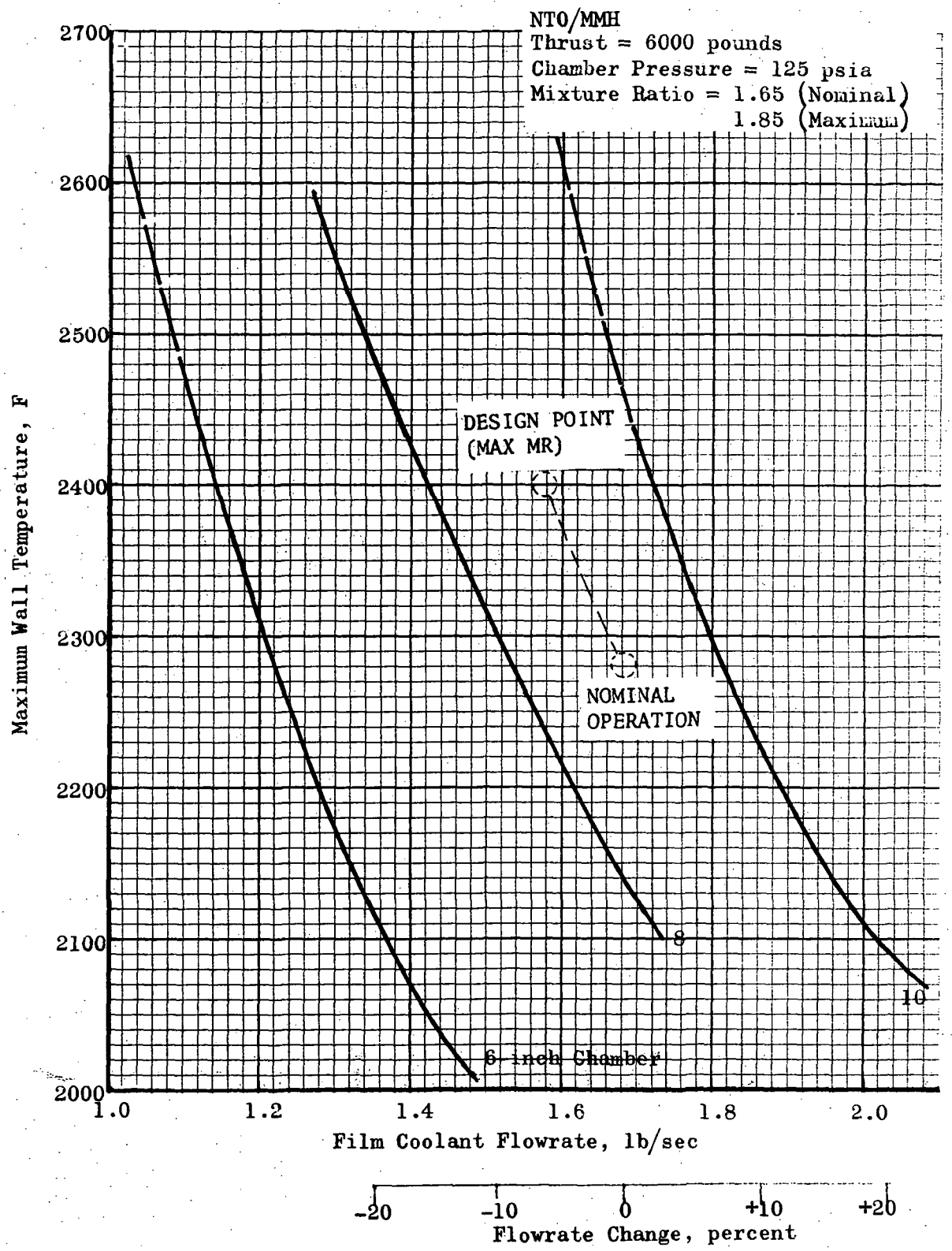


Figure 92. Sensitivity of Wall Temperature to Film Coolant Flow in Dump/Film Cooled Thrust Chambers

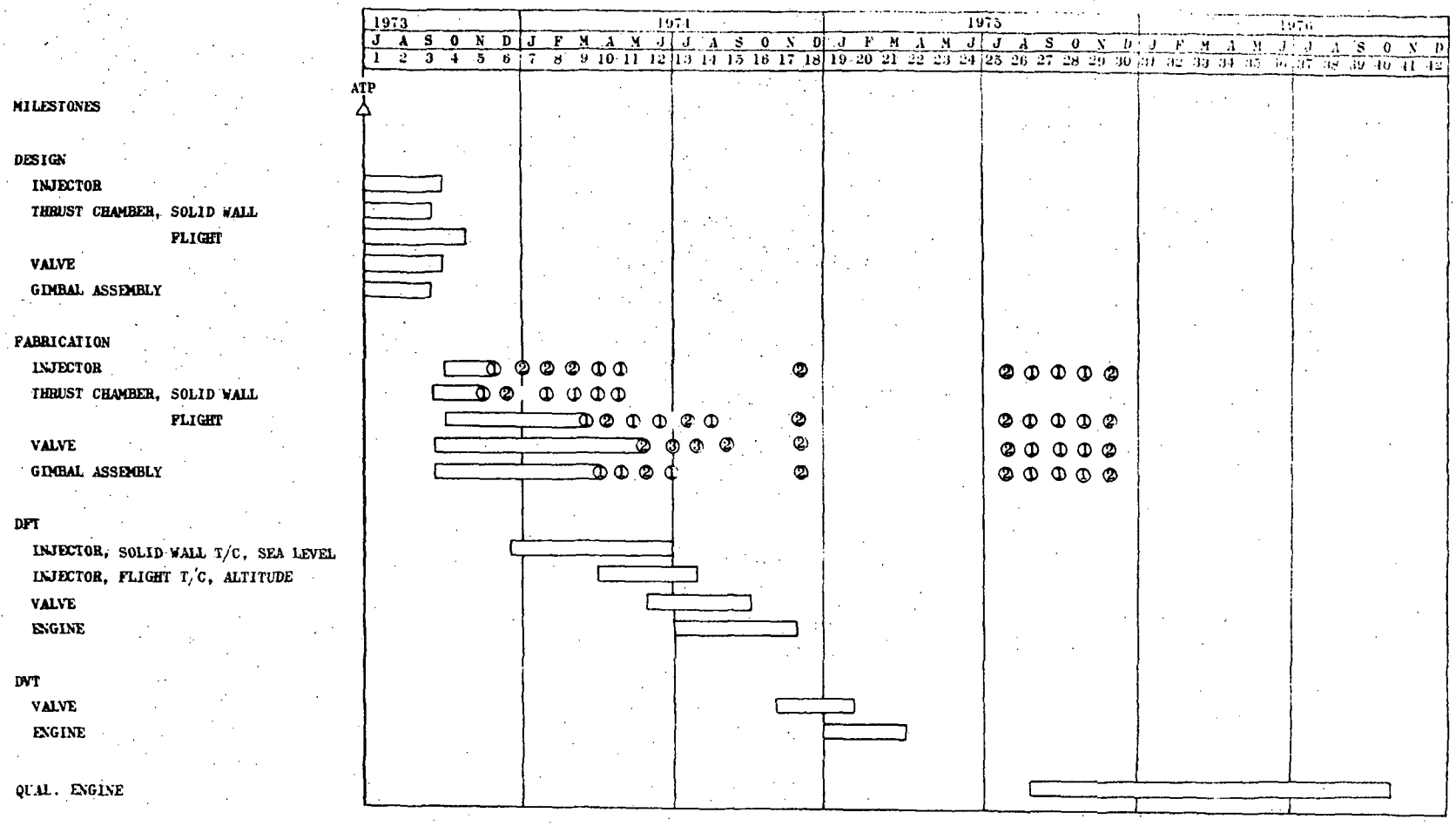


Figure 93. OME PROGRAM; NTO-MMH, NTO-50 50 with Regeneratively and Film Cooled Thrust Chamber

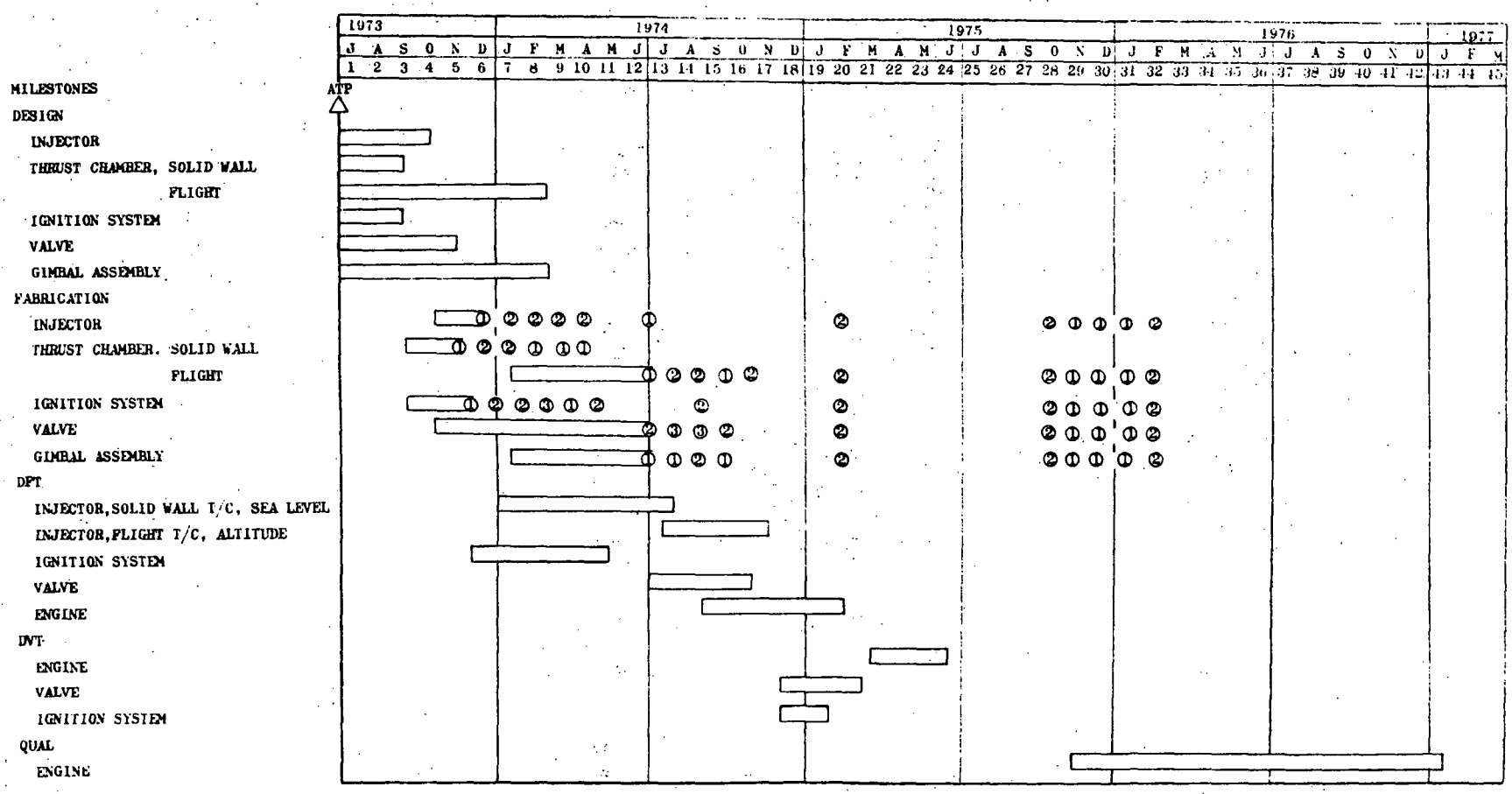


Figure 94. OME Program;LOX-MMH, LOX-50/50 with Regeneratively Cooled Thrust Chamber

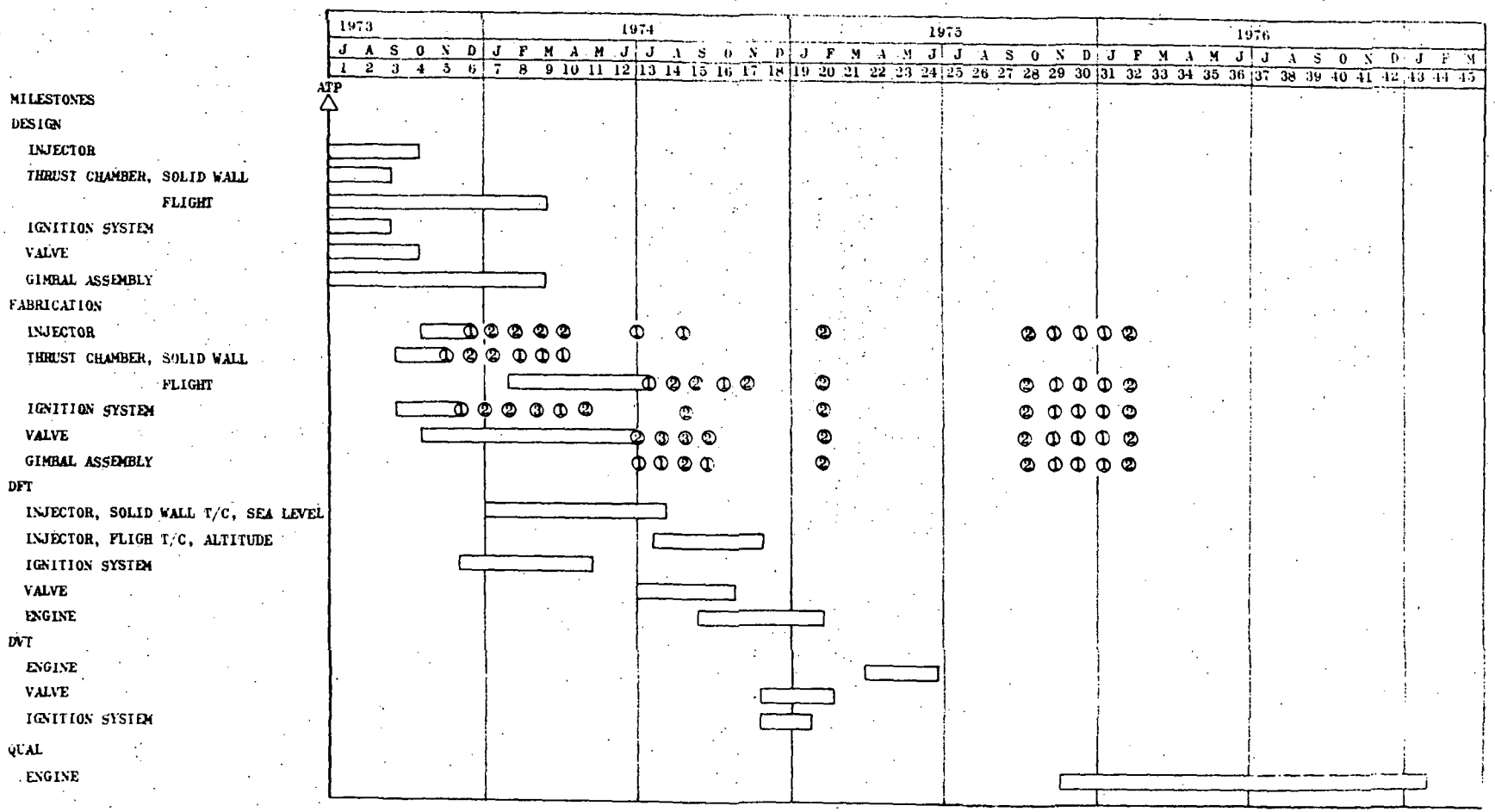


Figure 95. OME Program;LOX-MMH with Film Cooled Thrust Chamber

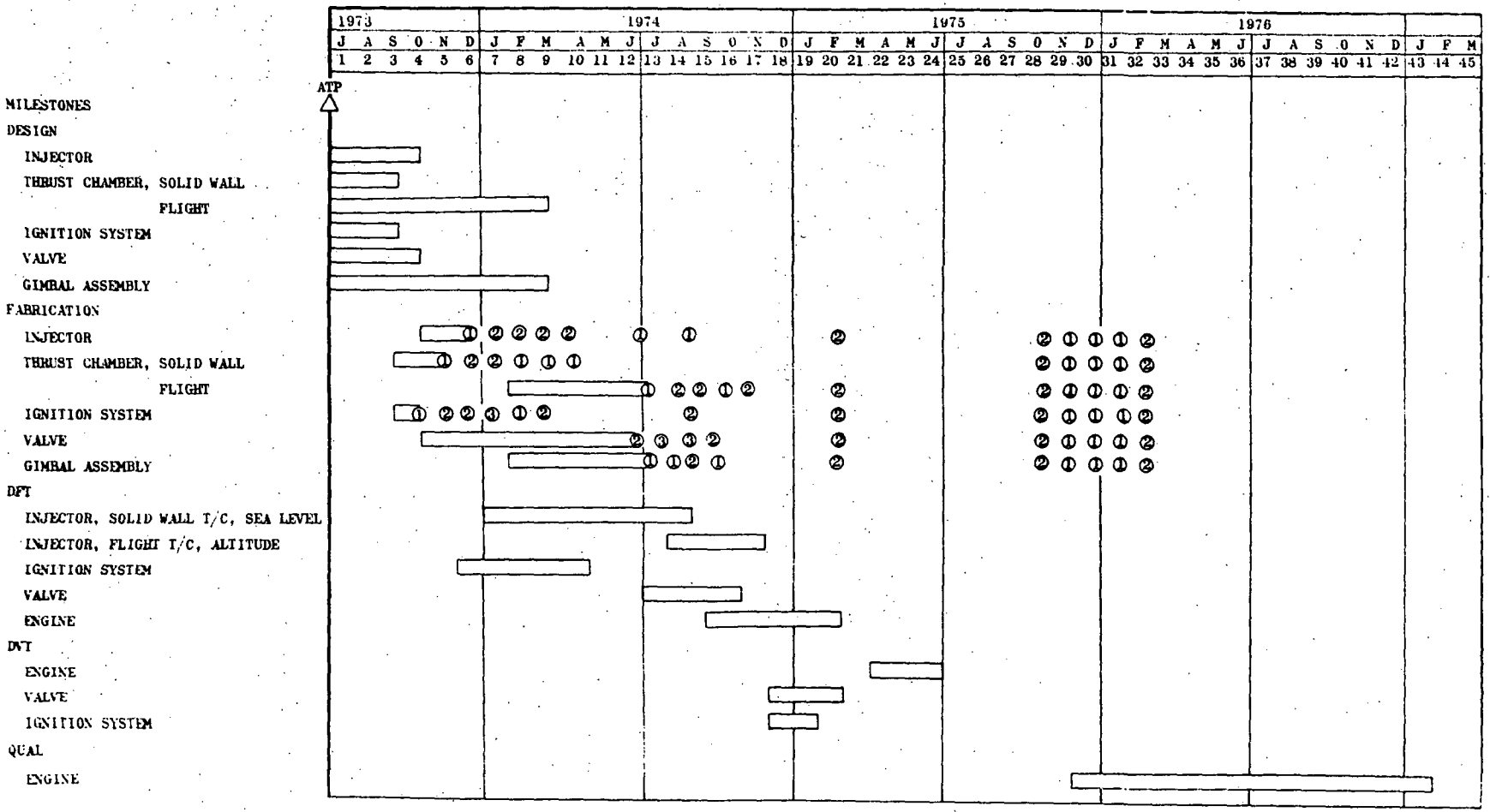


Figure 96. OME Program;LOX-RP with Regeneratively Cooled Thrust Chamber

APPENDIX A
THERMAL/LIFE ANALYSIS

REGENERATIVE COOLING ANALYSIS (Table A-1)

A radiation cooling analysis was conducted to determine the area ratio at which the radiation cooled nozzle can be attached (which was then defined as the area ratio at which the regenerative coolant enters the chamber). Allowable regenerative coolant bulk temperature rises were calculated based on a maximum bulk inlet temperature of 90 F. MMH and 50-50 were assumed to be allowed to heat up to within 50F of the boiling points based on chamber pressure. N_2H_4 was allowed to reach a maximum temperature of 280 F. Heat loads and corresponding bulk temperature rises were calculated for off-design conditions based on ± 10 percent chamber pressure and ± 12 percent o/f capabilities. The film coolant, if any, required to reduce the heat flux to meet the bulk temperature rise limits was then calculated. Film coollant requirements and associated performance penalties for various combustion chamber lengths were balanced against injector efficiency losses to optimize the combustion chamber length. Film coolant flowrates less than 2 percent of total flow were subsequently increased to 2 percent.

The coolant passage geometry was determined by first selecting the number and width of the channels based on fabrication considerations. The velocity required at each station in the chamber was determined from the empirical burnout heat flux correlations available for the various propellant combinations. The correlations used for 50-50 and MMH are shown in Fig. A-1 and A-2 . The corresponding channel area and height profiles were calculated from these velocity requirements. If the velocity requirement was so low as to dictate an unreasonably high channel height, the height was limited to prevent excessive thrust chamber weight. Resulting pressure and wall temperature profiles were then calculated. A safety factor of 2 at nominal design conditions was applied to the burnout heat fluxes for all propellant combinations. Final heat flux profiles and bulk temperature rises were calculated for the actual wall temperature profiles. Exit bulk temperature, cooling jacket pressure drop, and 3-dimensional wall temperature profiles were calculated for this final design, and used to predict a chamber life expectancy.

FILM COOLANT ANALYSIS (Table A-2)

Heat transfer coefficients were first calculated by boundary layer analysis for the case of no film coolant. These coefficients were used to determine the length of the duct which will allow the film coolant to be heated up to its bulk temperature limit. Heat transfer analyses were then conducted along the chamber wall in the liquid region, vapor region, and decomposition region for the film cooled portion of the chamber. The chamber section analyzed included the insulated portion and extended into the radiation cooled portion. The amount of film coolant was adjusted until the maximum wall temperature, which occurs at the end of the insulated wall region, did not exceed the maximum allowable value of 2400 F at off-design conditions. The resultant temperature distributions were used to evaluate the chamber life. Pressure drops in the duct were also calculated and determined to be lower than injector pressure drops with which they are in parallel.

STRUCTURAL AND LIFE ANALYSIS Table A-3; Fig. A-3, A-4)

The finite element method was used to evaluate structurally all critical areas with respect to stresses generated by pressures, forces and/or thermal gradients. The minimum safety factors used are tabulated on page

The basic approach was to determine the life of the thrust chamber based on combined effects of fatigue and creep as shown schematically in Fig. A-3 If creep and fatigue were both significant, then the total damage fraction was obtained by adding the creep damage fraction and the fatigue damage fraction with appropriate safety factors on each. The total damage fraction must be less than 1.0 for the chamber to be able to operate for the required number of hours and cycles.

Experimental data and the method of universal slopes were used to predict the number of cycles to failure on a fatigue basis (Fig. A-5 and A-6). This value was computed based on the thermal and pressure strains encountered during a cycle of operation at the design conditions. The fatigue damage fraction was computed by dividing the required number of

operational cycles by the predicted number of cycles to failure. The most severe creep conditions in the thrust chamber were evaluated to determine the time to rupture under these conditions. The creep damage fraction was determined by dividing the required operational time of the chamber by the time to rupture.

TABLE A-1. ANALYTICAL TECHNIQUES

REGENERATIVE COOLING

- SUPPLEMENTAL FILM COOLANT REQUIREMENTS
  - DETERMINE ϵ_{ATTACH} FOR RADIATION NOZZLE
  - DETERMINE ALLOWABLE ΔT_{BULK}
    - T_{IN} = 90F
    - $T_{OUT} = T_{SAT,\ P_C}$ -50F
    - BASED ON ± 10 PERCENT P_C AND ± 12 PERCENT O/F OFF-DESIGN CAPABILITY
  - CALCULATE HEAT FLUX PROFILE VS FILM COOLANT FLOW AND L_C
  - OPTIMIZE L_C

TABLE A-1. ANALYTICAL TECHNIQUES (Continued)

REGENERATIVE COOLING

- COOLING PASSAGE GEOMETRY
  - SELECT NUMBER AND WIDTH OF CHANNELS
  - CALCULATE REQUIRED VELOCITY PROFILE
  - CALCULATE CHANNEL HEIGHT PROFILE
  - CALCULATE PRESSURE DROP
  - CALCULATE WALL TEMPERATURE PROFILES
  - CALCULATE FINAL HEAT FLUX PROFILE AND ΔT_B
  - EVALUATE FINAL DESIGN FOR T_{WALL}, LIFE, T_B, ΔP

REGENERATIVE COOLING DESIGN PROGRAM

TABLE A-2. ANALYTICAL TECHNIQUES

FILM COOLING

- CALCULATE DRY WALL FILM COEFFICIENTS BY BOUNDARY LAYER ANALYSIS
- FOR GIVEN $\dot{W}_{FILM}$ DETERMINE DUCT LENGTH FOR $T_{BULK} = T_{SAT} - 50F$ (OFF-DESIGN)
- DETERMINE LIQUID LENGTH
- CALCULATE T_{FILM} VS DISTANCE FOR VAPOR
- DETERMINE T_{WALL} VS DISTANCE FOR INSULATED WALL
- DETERMINE T_{WALL} VS DISTANCE FOR RADIATING WALL
- REPEAT UNTIL T_{WALL} DOES NOT EXCEED MAXIMUM ALLOWABLE AND $\epsilon_{RADIATION} > 3$
- SIZE CHANNELS IN DUCT
- VERIFY LIFE

TABLE A-3. ANALYTICAL TECHNIQUES

STRUCTURAL ANALYSIS

- FINITE ELEMENT METHOD
- PRESSURES, FORCES, THERMAL GRADIENTS
- EQUIVALENT UNIAXIAL STRESS AND STRAIN
- BASIC MINIMUM SAFETY FACTORS
  - YIELD 1.1
  - ULTIMATE 1.4
  - FATIGUE 4
  - CREEP 4

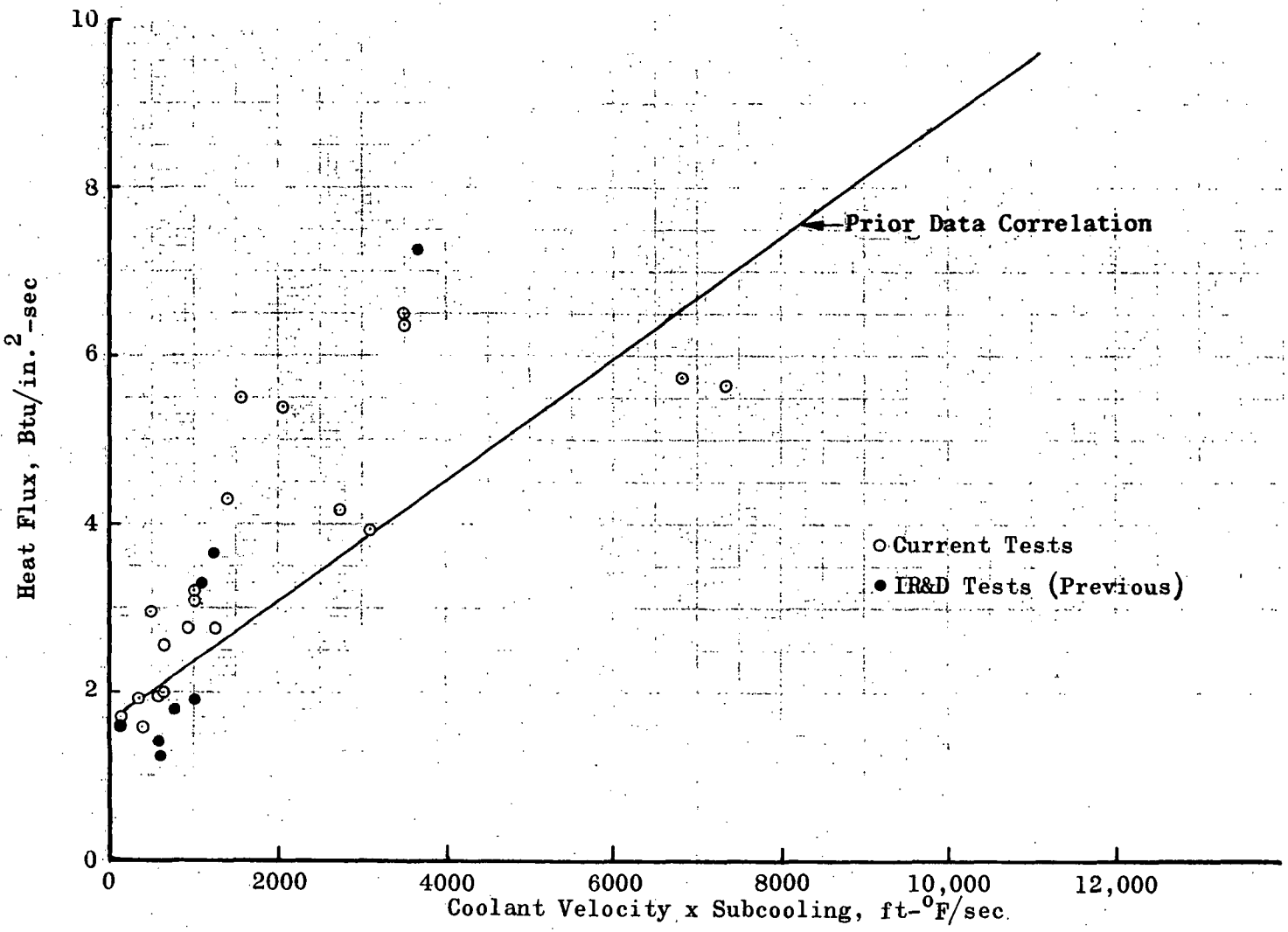


Figure A-1. Experimental Burnout Heat Flux Correlation for 50-50

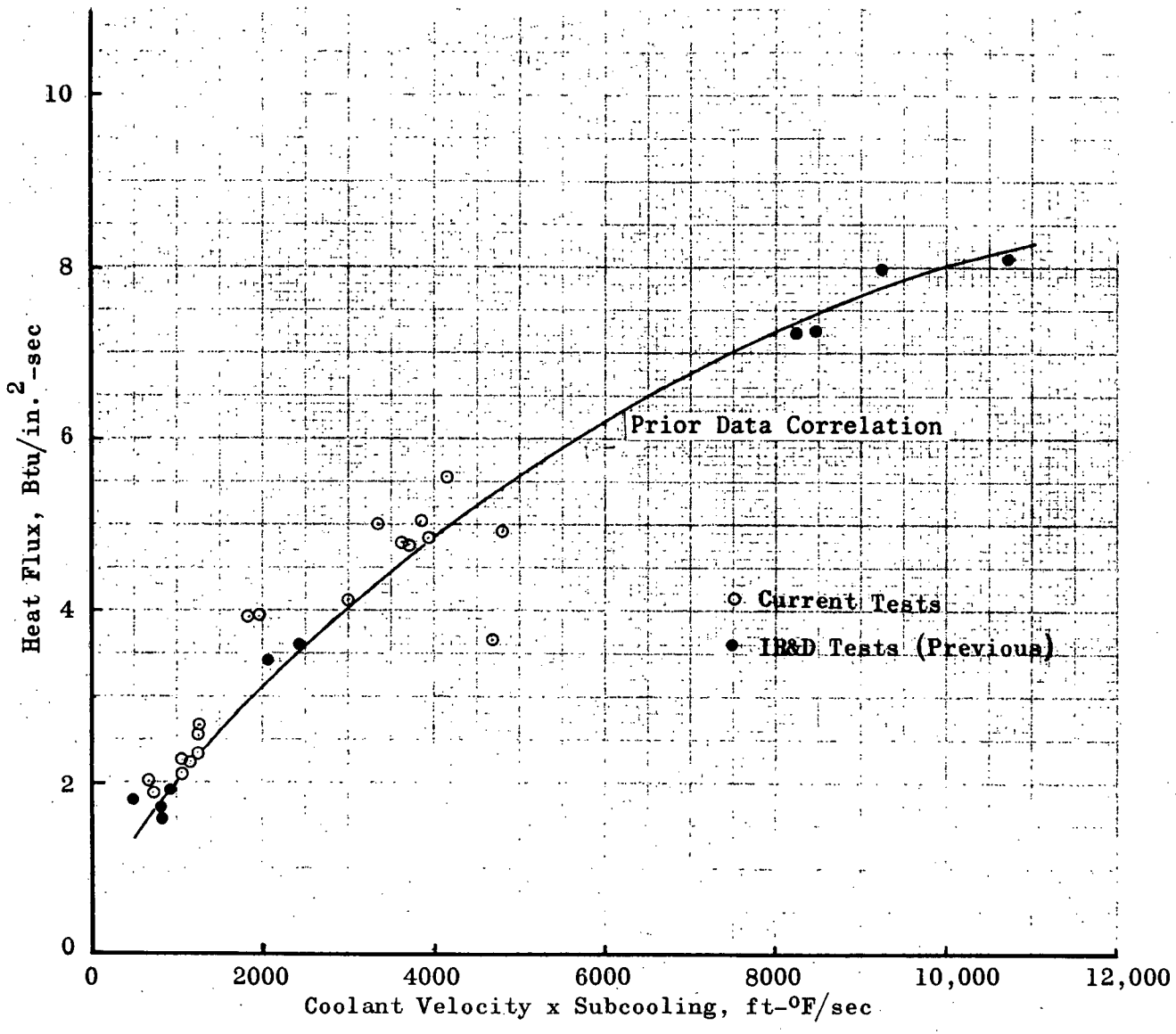


Figure A-2 . Experimental Burnout Heat Flux Correlation for MMH

Figure A-3. LIFE ANALYSIS

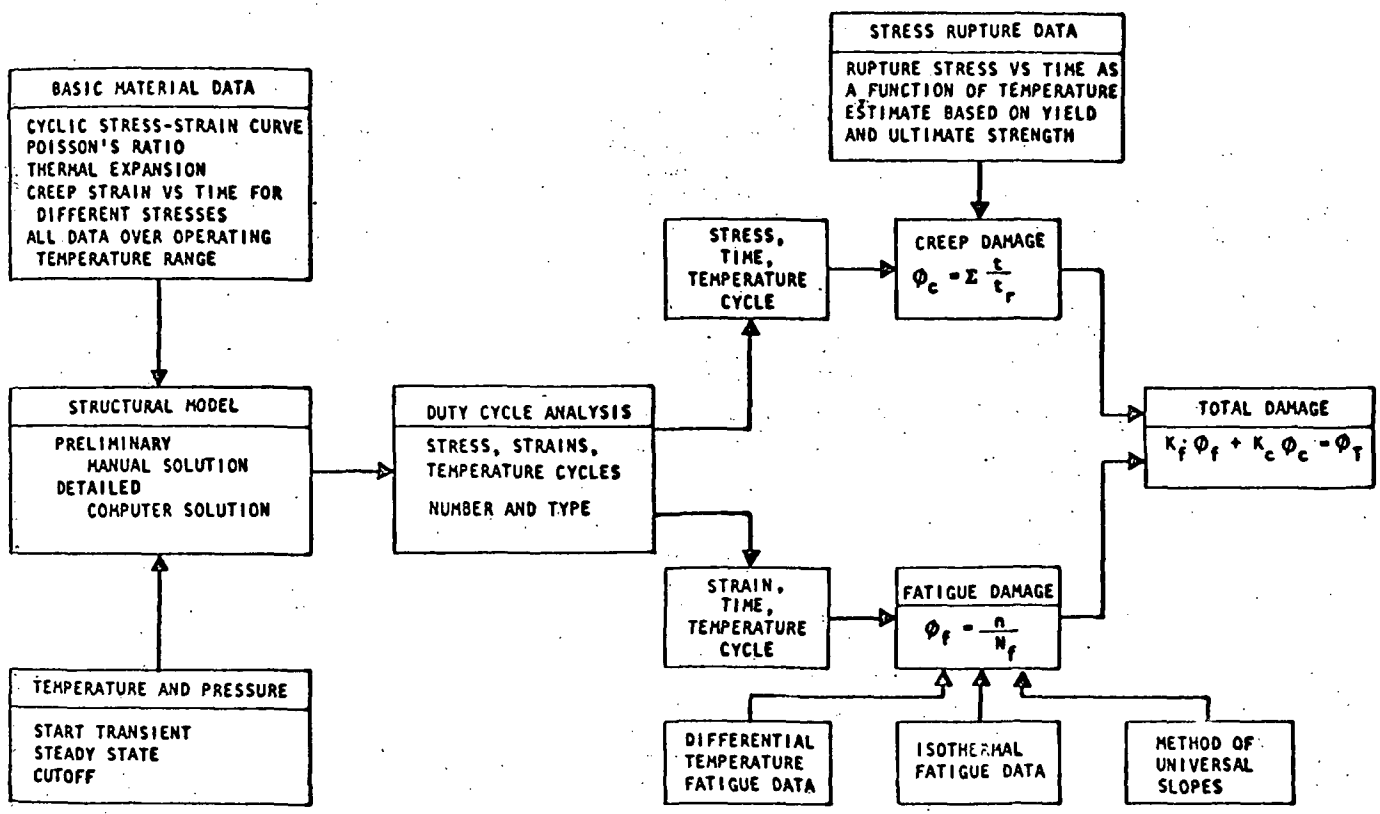


Figure A-4

LIFE ANALYSIS FOR DUMP/FILM COOLED CHAMBER

- LIFE FACTORS - CREEP AND FATIGUE
- CREEP DEPENDS ON T_{WALL} AND STRESS
- CREEP MOST SEVERE IN CONVERGING SECTION
- TWO MOST SEVERE FATIGUE REGIONS:
  - THROAT DURING START TRANSIENT
  - NEAR SLEEVE DURING STEADY STATE
- CREEP DAMAGE FRACTION $\approx$ 0.75
- FATIGUE DAMAGE FRACTION $\approx$ 0.05

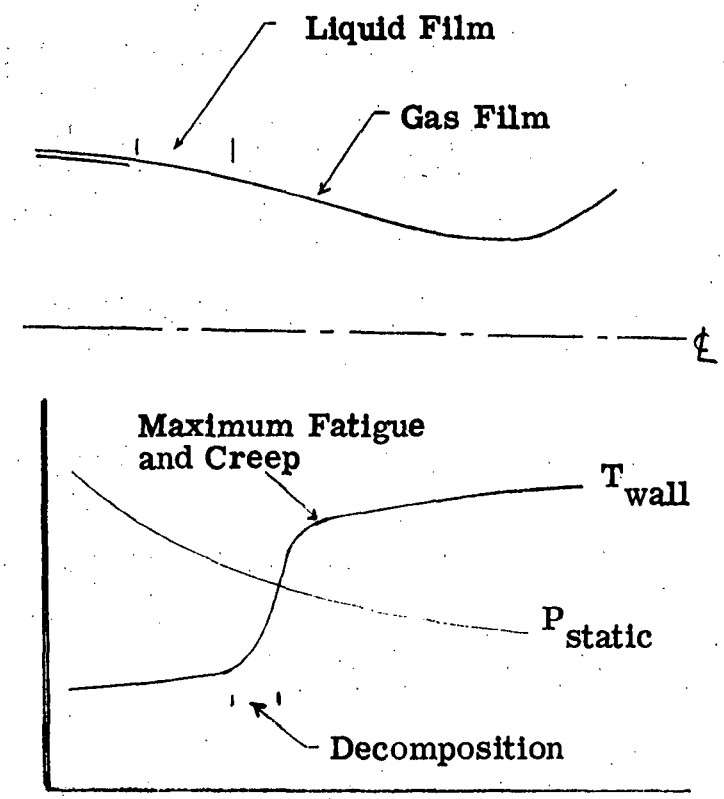


Figure A-5

UNIVERSAL SLOPES EQUATION FATIGUE DATA

STAINLESS STEEL TYPE 321

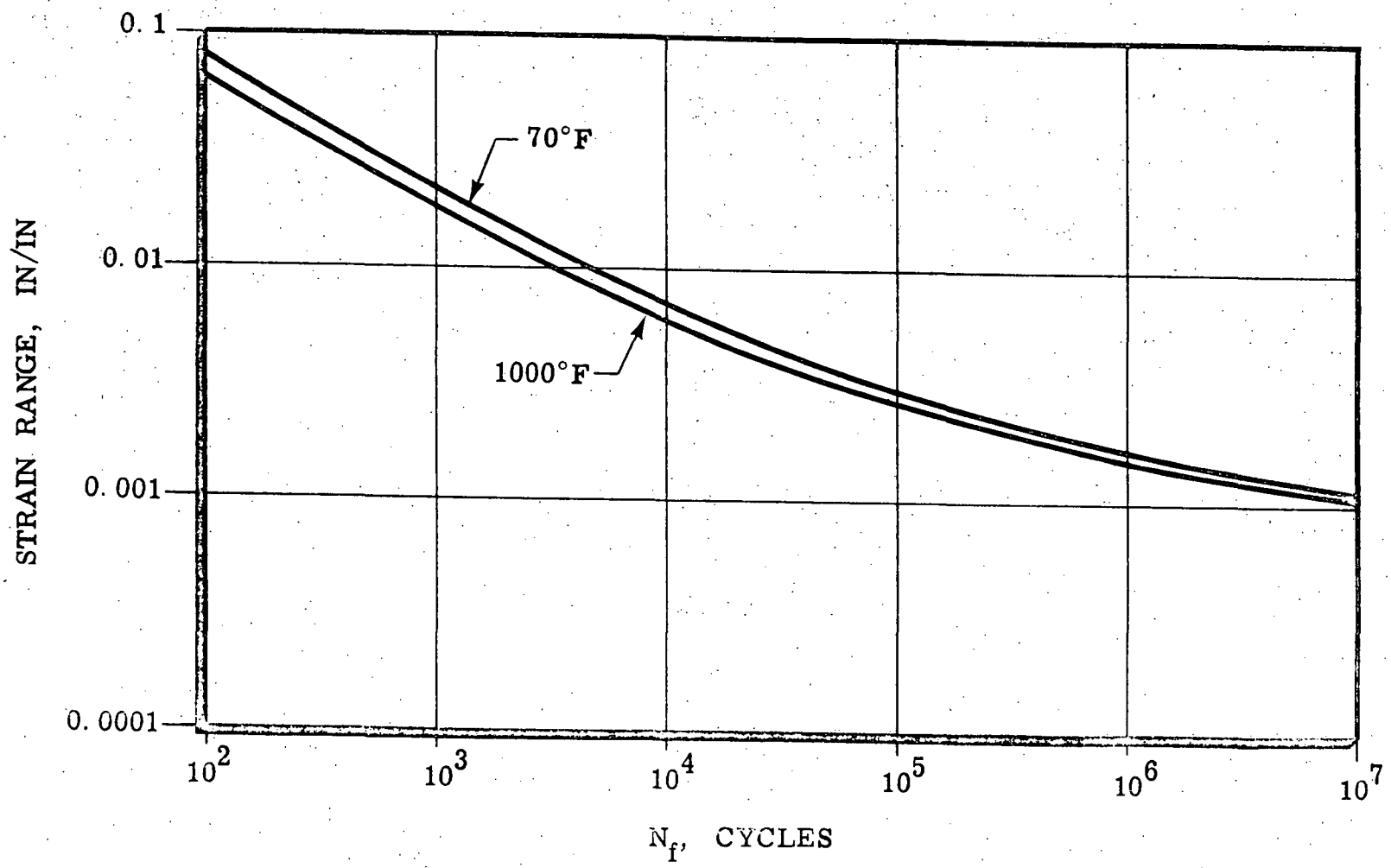
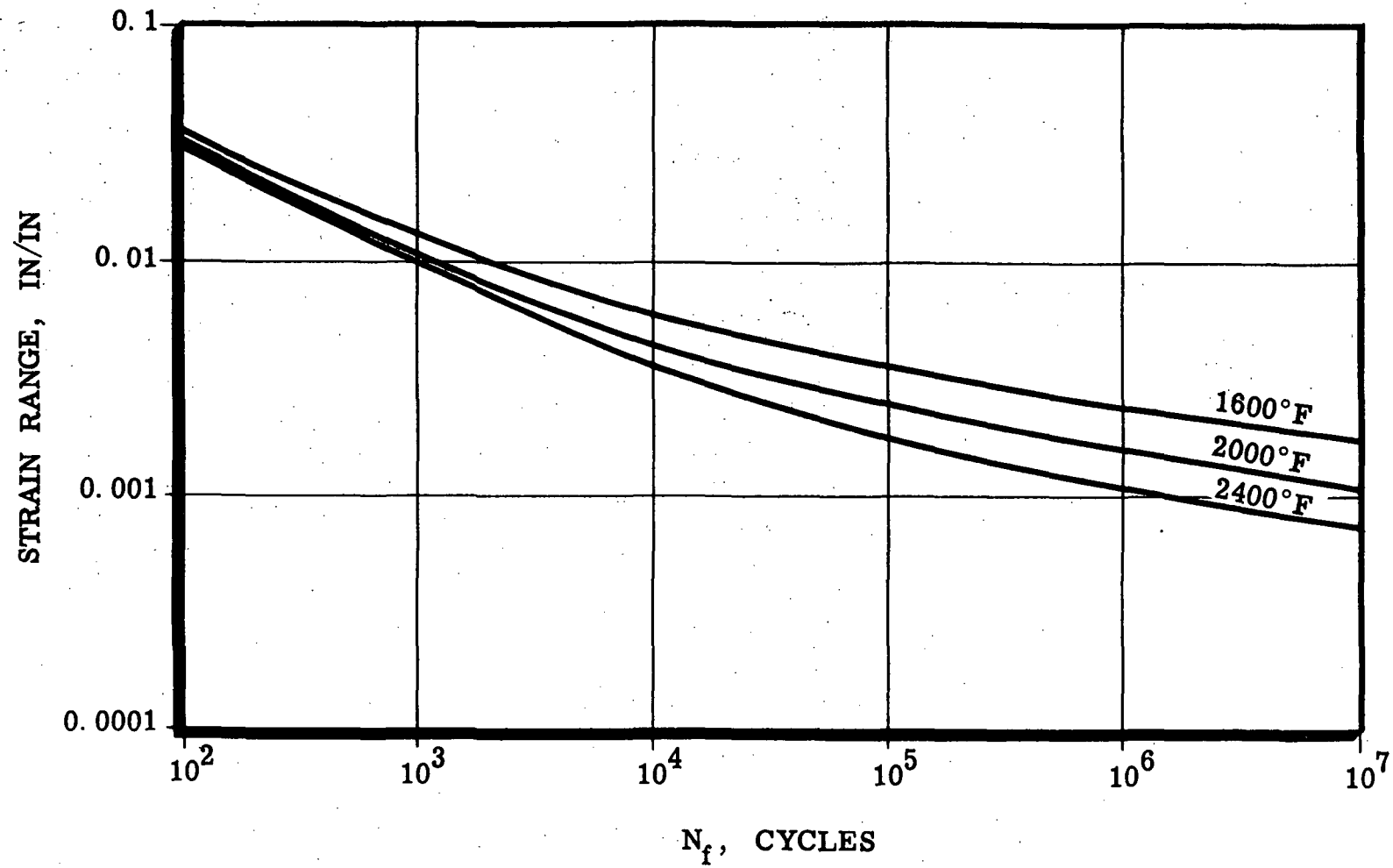



Figure A-6. UNIVERSAL SLOPES EQUATION FATIGUE DATA
COLUMBIUM ALLOY WC-103